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RESEARCH MEMORANDUM

ALTITUDE-WIND-TUNNEL INVESTIGATION OF PERFORMANCE CHARACTERISTICS OF A J47D PROTOTYPE (RX1-1) TURBOJET ENGINE WITH FIXED-AREA EXHAUST NOZZLE

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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ALTITUDE-WIND-TUNNEL INVESTIGATION OF PERFORMANCE CHARACTERISTICS OF A J47D PROTOTYPE (RX1-1) TURBOJET ENGINE WITH FIXED-AREA EXHAUST NOZZLE

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SUMMARY

An investigation has been conducted in the NACA Lewis altitude wind tunnel to determine the over-all performance of a prototype model of the J47D (RX1-1) turbojet engine operating with a fixed-area exhaust nozzle. Data were obtained for a range of engine speeds at altitudes from 5000 to 55,000 feet and flight Mach numbers from 0.18 to 0.71. The performance data were generalized by several methods to determine the range of flight conditions for which performance could be predicted from data obtained at a given flight condition.

Generalized engine performance data indicated that data obtained at a given altitude and flight Mach number could be used to predict net thrust for altitudes up to 55,000 feet at all corrected engine speeds, air flow for altitudes up to 45,000 feet with reasonable accuracy over most of the corrected engine speed range, and performance variables dependent on fuel flow for altitudes up to 35,000 feet with minimum error at high corrected engine speeds. Generalization of engine performance in terms of pumping characteristics indicated that data obtained at one flight condition could be used to predict jet thrust and specific fuel consumption at another flight condition within a relatively wide range of altitude, flight Mach number, and engine total-temperature ratios.

A minimum specific fuel consumption of 1.05 was obtained at an engine speed of 6600 rpm for altitudes from 6000 to 35,000 feet at a flight Mach number of 0.18. An increase in flight Mach number from 0.18 to 0.71 at an altitude of 25,000 feet raised the minimum specific fuel consumption from 1.05 to 1.27 and these values occurred at engine speeds of 6600 and 7300 rpm, respectively. The increase in exhaust-gas temperature and the resulting reduction in temperature-limited engine speed, which occurred with an increase in altitude, indicated the need for a variable-area exhaust nozzle for operation at rated engine speed at high altitudes and low flight Mach numbers.

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INTRODUCTION

An investigation was conducted in the NACA Lewis altitude wind tunnel to evaluate the performance of a J47D prototype (RX1-1) turbojet engine and its integrated electronic control with and without exhaust reheat under steady-state and transient operating conditions. As part of the over-all program, data on engine performance, component performance, and operational characteristics were obtained with fixed- and variable-area exhaust nozzles. The performance of a J47D (RX1-1) engine operating with a fixed-area exhaust nozzle is presented herein. 2108

The variation of engine performance variables with engine speed is shown graphically for simulated altitudes from 6000 to 55,000 feet at a flight Mach number of 0.18 and for flight Mach numbers from 0.18 to 0.71 at an altitude of 25,000 feet. Performance data are generalized to determine the suitability of correction factors for predicting engine performance over a range of flight conditions from data obtained at a given flight condition. Generalization in terms of engine pumping characteristics is also presented. All performance data obtained in this investigation are presented in tabular form.

APPARATUS

Engine

The J47D (RX1-1) engine used in the altitude-wind-tunnel investigation has no official manufacturer's rating; however it has a minimum sea-level static-thrust rating (with the afterburner not operating) of 5700 pounds at an engine speed of 7950 rpm and a turbine-outlet exhaust-gas temperature of 1275° F; at this rating the engine air flow is approximately 99 pounds per second The engine has a twelve-stage axial-flow compressor with a pressure ratio of about 5.1 at rated engine speed, eight cylindrical direct-flow-type combustion chambers, and a single-stage impulse turbine. For these tests a fixed-area exhaust nozzle was used. The exhaust nozzle used in this investigation has an outlet area of 285.5 square inches, which produces a turbine-outlet temperature of 1275° F at an altitude of 5000 feet, a flight Mach number of 0.18, and an engine speed of 7950 rpm. The over-all length of the engine without the exhaust nozzle is 143 inches, the maximum diameter is approximately 37 inches, and the total weight is 2475 pounds.

Installation

The engine was mounted on a wing in the tunnel test section (fig. 1). Dry refrigerated air was supplied to the engine from the tunnel make-up air system through a duct connected to the engine inlet. Engine thrust and drag measurements by the tunnel balance scales were made possible by a frictionless slip joint located in the duct upstream of the engine. The air flow through the duct was throttled from approximately sea-level pressure to a total pressure at the engine inlet corresponding to the desired flight Mach number at a given altitude.

Instrumentation for measuring pressures and temperatures was installed at various stations in the engine (fig. 2).

PROCEDURE

Engine performance data were obtained over a range of engine speeds at the following altitudes and flight Mach numbers:

Altitude (ft.)	Flight Mach number
5,000	0.18
6,000	.18
15,000	.18, .51
25,000	.18, .51, .71
35,000	.18
45,000	.18
55,000	.22

Complete ram pressure recovery at the compressor inlet was assumed in the calculation of flight Mach number. Engine inlet-air temperatures were held at approximately NACA standard values for each flight condition except for altitudes above 25,000 feet where the lowest engine inlet-air temperature obtained was about 436° R. Fuel conforming to specification MIL-F-5624 (AN-F-58a), with a lower heating value of 18,900 Btu per pound, was used throughout the investigation.

Thrust values were calculated from both the tunnel balance-scale measurements and from values of gas flow and jet velocity obtained from measurements by the exhaust-nozzle-outlet survey rake. The exhaust-nozzle jet coefficient, defined as the ratio of scale jet thrust to rake jet thrust, is presented as a function of exhaust-nozzle pressure ratio in figure 3. The engine performance presented herein is based on thrust values obtained from scale measurements inasmuch as this method includes the thrust losses resulting from the inefficiency of the exhaust nozzle. Symbols and methods of calculations are given in appendixes A and B, respectively.

RESULTS AND DISCUSSION

All the data obtained in the performance investigation of the engine are compiled in table I. Inasmuch as engine inlet-air temperatures below 436° R were not obtained and because small errors occurred in setting the tunnel static pressure, the data presented graphically in nongeneralized form have been adjusted to NACA standard altitude conditions by use of the factors δ_a and θ_a. (See appendix A.) 2108

Effect of altitude. - Engine performance data at altitudes from 6000 to 55,000 feet at a flight Mach number of approximately 0.18 are presented in figure 4 to show the effect of variations in altitude on net thrust, air flow, fuel flow, specific fuel consumption, fuel-air ratio, and exhaust-gas total temperature.

As the altitude was increased, engine net thrust, air flow, and fuel flow decreased (figs. 4(a) to 4(c)). The specific fuel consumption was not significantly affected by a change in altitude from 6000 to 35,000 feet at engine speeds above 6200 rpm (fig. 4(d)). A minimum specific fuel consumption of 1.05 pounds of fuel per pound of net thrust was obtained at an engine speed of about 6600 rpm for altitudes from 6000 to 35,000 feet. At an altitude of 55,000 feet, the minimum specific fuel consumption increased to 1.27 and occurred at an engine speed of 6800 rpm. This increase in specific fuel consumption is attributed to a reduction in component efficiencies and partly to the higher flight Mach number at which data were obtained at an altitude of 55,000 feet. In general, the fuel-air ratio increased with an increase in altitude (fig. 4(e)).

The exhaust-gas total temperature (fig. 4(f)) was not greatly affected by an increase in altitude from 6000 to 25,000 feet at engine speeds above approximately 7200 rpm. The slope of the temperature curve increased with a change in altitude from 6000 to 35,000 feet, however, so that the temperature generally tended to increase at high engine speeds and decrease at low engine speeds as altitude was increased. A further increase in altitude from 35,000 to 55,000 feet resulted in an increase in exhaust-gas total temperature at each engine speed. Inasmuch as engine-inlet temperatures were higher than for NACA standard altitude conditions at the higher altitudes, the adjusted exhaust-gas temperatures do not extend to the limiting temperature line. Extrapolation of the data indicates, however, that an increase in altitude from 6000 to 25,000 feet would reduce the temperature-limited engine speed from approximately 7920 to 7780, whereas a further increase in altitude to 55,000 feet would reduce the temperature-limited speed to about 7100 rpm. Obviously at high altitudes and low flight Mach numbers a variable-area exhaust nozzle is required in order to maintain rated engine speed without exceeding present exhaust-gas temperature limits.

Effect of flight Mach number. - Engine performance data for flight Mach numbers from 0.18 to 0.71 at an altitude of 25,000 feet are presented in figure 5 to show the effect of variations in flight Mach
2108 number on engine net thrust, air flow, fuel flow, specific fuel consumption, fuel-air ratio, and exhaust-gas total temperature.

At low engine speeds, the net thrust decreased with an increase in flight Mach number (fig. 5(a)). The rate of increase of net thrust with engine speed became greater, however, as flight Mach number was raised so that at high engine speeds the net thrust increased with flight Mach number. The engine air flow (fig. 5(b)) increased with an increase in flight Mach number at all engine speeds. An increase in flight Mach number reduced the engine fuel flow (fig. 5(c)) at engine speeds below 6000 rpm and increased the fuel flow at higher engine speeds. Specific fuel consumption (fig. 5(d)) increased with an increase in flight Mach number at all engine speeds. The minimum specific fuel consumption increased from 1.05 at a flight Mach number of 0.18 to 1.22 at a flight Mach number of 0.51 and occurred at engine speeds of 6600 and 7000 rpm, respectively. A further increase in flight Mach number to 0.71 increased the minimum specific fuel consumption to 1.27 and occurred at an engine speed of 7300 rpm. Extrapolation of the data indicates that at temperature-limited engine speed, an increase in flight Mach number from 0.18 to 0.51 would increase the specific fuel consumption from about 1.15 to 1.30, whereas a further increase in flight Mach number to 0.71 would raise the specific fuel consumption to about 1.32. Engine fuel-air ratio (fig. 5(e)) was reduced at all engine speeds by an increase in flight Mach number. The exhaust-gas total temperature (fig. 5(f)) decreased with an increase in flight Mach number at all engine speeds but the effect was small in the high engine-speed range. The temperature-limited engine speed increased from 7850 rpm at a flight Mach number of 0.51 to 7920 rpm at a flight Mach number of 0.71.

Generalized performance. - Performance data for altitudes from 6000 to 55,000 feet and a flight Mach number of approximately 0.18 have been generalized to standard sea-level conditions by use of the correction factors δ and θ. (See appendix A.) The derivation of these factors (reference 1) does not account for the effect of flight Mach number or for changes in component efficiencies such as those associated with variations in Reynolds numbers. Consequently, any changes in flight Mach number or component efficiencies lessen the possibility of defining engine performance variables obtained at various altitudes by a single curve.

Engine performance data obtained at altitudes from 6000 to 55,000 feet and a flight Mach number of approximately 0.18 are presented in figure 6 to show the effect of altitude on the relation between corrected engine speed and corrected values of net thrust, air flow, fuel flow, specific fuel consumption, fuel-air ratio, and exhaust-gas total temperature.

Corrected net thrust (fig. 6(a)) reduced to a single curve for the entire range of altitudes and corrected engine speeds investigated. The corrected engine air flows (fig. 6(b)) formed a single curve for altitudes up to 45,000 feet at engine speeds up to 6300 rpm and decreased with an increase in altitude above 15,000 feet at higher engine speeds. Corrected air flows at an altitude of 55,000 feet were scattered and were inconsistent with the other altitudes because of small variations in flight Mach number from one engine speed to another and because the average flight Mach number was higher than that for the data obtained at the other altitudes. Corrected fuel flow (fig. 6(c)), corrected specific fuel consumption (fig. 6(d)), corrected fuel-air ratio (fig. 6(e)), and corrected exhaust-gas total temperature (fig. 6(f)) formed a single curve for altitudes of 6000 and 15,000 feet and also for altitudes of 25,000 and 35,000 feet over most of the range of corrected engine speeds. With these exceptions, each of the generalized variables dependent on fuel flow increased with an increase in altitude, which indicates a reduction in engine component efficiencies. Thus, a generalization of individual performance variables indicates that data obtained at a given altitude and flight Mach number could be used to predict (1) net thrust for altitudes up to 55,000 feet at all corrected engine speeds, (2) air flows for altitudes up to 45,000 feet with reasonable accuracy over most of the engine-speed range, and (3) fuel-flow and performance variables dependent on fuel flow for altitudes up to 35,000 feet with minimum error at high corrected engine speeds.

Generalization in terms of pumping characteristics. - Engine performance may be generalized in terms of the over-all engine total-temperature ratio and total-pressure ratio, which define the over-all change in available energy of the air flowing through the engine. Changes in component efficiencies with altitude lessen the possibility of reducing data to a single curve.

Within the range of flight conditions where the relation between engine total-pressure ratio and engine total-temperature ratio is defined by a single line, data obtained at one flight condition can be used to determine the exhaust-gas total pressure at another flight condition for a given value of exhaust-gas total temperature. Consequently, jet thrust can be calculated from equation (7) or (9) (appendix B).

The variation of engine total-temperature ratio with engine total-pressure ratio is shown in figure 7(a) for altitudes from 6000 to 55,000 feet at a flight Mach number of approximately 0.18 and in figure 7(b) for flight Mach numbers from 0.18 to 0.71 at an altitude of 25,000 feet. Engine total-temperature ratios formed a single curve for all engine pressure ratios investigated at altitudes from 6000 to 35,000 feet. An increase in altitude above 35,000 feet increased the total-temperature ratio at each value of total-pressure ratio (fig. 7(a)). Engine total-temperature ratios for flight Mach numbers from 0.18 to 0.71 formed a single curve at engine temperature ratios above 2.30 (fig. 7(b)). Thus, data obtained at one flight condition can be used to predict jet thrust at another flight condition within the following ranges of operating conditions: (1) altitudes up to 25,000 feet at flight Mach numbers from 0.18 to 0.71 and engine total-temperature ratios above 2.30, and (2) altitudes up to 35,000 feet at a flight Mach number of 0.18 and engine total-temperature ratios above 2.80. (Data were not obtained at Mach numbers above 0.18 or temperature ratios below 2.80 at an altitude of 35,000 ft.)

Another method of presenting engine pumping characteristics is shown in figure 8 where the engine total-pressure and total-temperature ratios are plotted as functions of corrected fuel flow for altitudes from 6000 to 55,000 feet at a flight Mach number of approximately 0.18 (fig. 8(a)) and for flight Mach numbers from 0.18 to 0.71 at an altitude of 25,000 feet (fig. 8(b)). In order to account for the rise in total pressure and temperature at the compressor inlet with an increase in flight Mach number and thereby eliminate the dispersion of data obtained at different flight Mach numbers, the fuel flow was corrected by the factors δ_T and θ_T, which are based on total pressure and total temperature at the compressor inlet, respectively, and are defined in appendix A. Predictions of engine performance from one flight condition to another are valid only within the range of flight and engine operating conditions at which both the total-pressure and total-temperature ratios form a single line.

Thus, the data presented in figures 8(a) and 8(b) indicate that the jet thrust and specific fuel consumption can be predicted within the following ranges of operating conditions: (1) altitudes up to 25,000 feet at flight Mach numbers from 0.18 to 0.71 and engine total-temperature ratios above 2.30, (2) altitudes up to 25,000 feet at flight Mach numbers from 0.51 to 0.71 and engine total-temperature ratios above 2.00, and (3) altitudes up to 35,000 feet at a flight Mach number of 0.18 and engine total-temperature ratios above 2.80. The limitations imposed on the third operating range result from the lack of data to substantiate the validity of performance predictions at higher flight Mach numbers and lower engine total-temperature ratios. The reductions in total-pressure and -temperature ratios for constant fuel flows at altitudes above 35,000 feet can be attributed to the reduction in component efficiencies associated primarily with Reynolds number effects.

It is of interest to note that for the range of altitudes investigated the correlation of engine total-temperature ratio plotted as a function of corrected fuel flow (fig. 8(a)) was better than the correlation of either corrected fuel flow or corrected exhaust-gas temperature plotted as functions of corrected engine speed (figs. 6(c) and 6(f), respectively). This phenomenon apparently resulted from simultaneous reductions in component efficiencies as altitude was increased in that the corrected exhaust-gas temperature increased with a reduction in compressor efficiency whereas the corrected fuel flow increased with a reduction in both compressor and combustion efficiency. The combined effects of these changes were such as to maintain good correlation in terms of pumping characteristics.

SUMMARY OF RESULTS

The following results were obtained from the altitude wind tunnel investigation of the J47D prototype (RX1-1) turbojet engine operating with a fixed-area exhaust nozzle at simulated altitudes from 6000 to 55,000 feet for flight Mach numbers from 0.18 to 0.71:

1. Generalized engine performance data indicated that data obtained at a given altitude and flight Mach number could be used to predict net thrust for altitudes up to 55,000 feet at all operable corrected engine speeds. Air flow could be predicted with reasonable accuracy for altitudes up to 45,000 feet over most of the corrected engine speed range. Performance variables dependent on fuel flow could be predicted for altitudes up to 35,000 feet with minimum error at high corrected engine speeds.

2. From engine pumping characteristics obtained at a given altitude and flight Mach number, the jet thrust and specific fuel consumption could be predicted within the following ranges of operation conditions: altitudes up to 25,000 feet at flight Mach numbers from 0.18 to 0.71 and engine total-temperature ratios above 2.30; altitudes up to 25,000 feet at flight Mach numbers from 0.51 to 0.71 and engine total-temperature ratios above 2.00; and altitudes up to 35,000 feet at a flight Mach number of 0.18 and engine total-temperature ratios above 2.80.

3. Minimum specific fuel consumption of 1.05 was obtained at engine speed of about 6600 rpm at altitudes from 6000 to 35,000 feet at a flight Mach number of 0.18. An increase in flight Mach numbers from 0.18 to 0.71 at an altitude of 25,000 feet increased the minimum specific fuel consumption from 1.05 to 1.27, which were obtained at engine speeds of 6600 and 7300 rpm, respectively.

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4. At high engine speeds, an increase in altitude increased the exhaust-gas temperature, indicating a reduction in temperature-limited engine speed and the need for a variable-area exhaust nozzle for operation at rated engine speed at high altitudes and low flight Mach numbers.

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APPENDIX A

SYMBOLS

The following symbols were used on the figures and calculations: 2108

A	cross-sectional area, sq ft
B	thrust scale reading, lb
C_j	exhaust-nozzle jet coefficient
C_T	ratio of hot exhaust-nozzle area to cold exhaust-nozzle area
D	external drag of installation, lb
D_r	exhaust-nozzle tail-rake drag, lb
F_j	jet thrust, lb
F_n	net thrust, lb
f/a	fuel-air ratio
g	acceleration due to gravity, 32.2 ft/sec^2
P	total pressure, lb/sq ft absolute
p	static pressure, lb/sq ft absolute
M_0	flight Mach number
N	engine speed, rpm
R	gas constant, 53.3 ft-lb/(lb)($^{\circ}$R)
T	total temperature, $^{\circ}$R
T_i	indicated temperature, $^{\circ}$R
t	static temperature, $^{\circ}$R
V	velocity, ft/sec
W_a	air flow, lb/sec

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W_f fuel flow, lb/hr

W_f/F_n specific fuel consumption, lb/(hr)(lb net thrust)

γ ratio of specific heats

δ ratio of tunnel static pressure (p_0) to the absolute static pressure of NACA standard atmosphere at sea level

δ_a ratio of tunnel static pressure (p_0) to the absolute static pressure of NACA standard altitude

δ_T ratio of total pressure at compressor inlet to absolute static pressure of NACA standard atmosphere at sea level

θ ratio of absolute equivalent ambient static temperature to absolute static temperature of NACA standard atmosphere at sea level

θ_a ratio of absolute equivalent ambient static temperature to absolute static temperature of NACA standard altitude

θ_T ratio of absolute total temperature at compressor inlet to absolute static temperature of NACA standard atmosphere at sea level

Subscripts:

0 free air stream

1 engine inlet

6 turbine outlet

7 1-in. upstream of exhaust-nozzle outlet

e equivalent

r rake

s scale

x inlet duct 6 in. upstream of frictionless slip-joint flange

y inlet duct $28\frac{3}{4}$ in. downstream of frictionless slip-joint flange

APPENDIX B

METHODS OF CALCULATION

Flight Mach number. - The flight Mach number assuming complete ram pressure recovery was computed as

$$M_0 = \sqrt{\frac{2}{\gamma_1 - 1}\left[\left(\frac{P_1}{p_0}\right)^{\frac{\gamma_1 - 1}{\gamma_1}} - 1\right]} \qquad (1)$$

Temperature. - Total temperature was determined by using a calibrated thermocouple with impact-recovery factor of 0.85 from the indicated temperature by

$$T = \frac{T_i\left(\frac{P}{p}\right)^{\frac{\gamma-1}{\gamma}}}{1 + 0.85\left[\left(\frac{P}{p}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]} \qquad (2)$$

Equivalent temperature. - Equivalent temperature was obtained from the adiabatic relation of pressures and temperatures,

$$t_e = \frac{T_1}{\left(\frac{P_1}{p_0}\right)^{\frac{\gamma_1 - 1}{\gamma_1}}} \qquad (3)$$

Engine air flow. - The engine air flow was determined from measurements at the engine inlet (station 1), by

$$W_{a,1} = A_1 p_1 \sqrt{\left(\frac{2\gamma_1}{\gamma_1 - 1}\right)\frac{g}{t_1 R}\left[\left(\frac{P_1}{p_1}\right)^{\frac{\gamma_1 - 1}{\gamma_1}} - 1\right]} \qquad (4)$$

Thrust. - The thrust was obtained from two sources: (1) the balance-scale measurements; and (2) the temperature and the pressure measured at the nozzle outlet (station 7).

Jet thrust determined from the balance-scale measurements was calculated from the equation

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$$F_{j,s} = D + B + D_r + \frac{W_{a,1} V_y}{g} + A_x \ (p_x - p_0) \tag{5}$$

The drag of the engine installation D was determined with the engine inoperative and with a blind flange installed at the engine inlet to prevent air flow through the engine. The rake drag D_r was measured by a pneumatic balance piston mechanism. The last two terms in equation (5) represent the momentum and pressure forces acting on the installation at the slip joint in the inlet-air duct.

The net thrust was obtained by subtracting the equivalent momentum of the air at the engine inlet from the jet thrust

$$F_{n,s} = F_{j,s} - \frac{W_{a,1} V_e}{g} \tag{6}$$

The ideal or rake jet thrust based on a survey at the exhaust-nozzle outlet, was obtained from the equation

$$F_{j,r} = \frac{2\gamma_7}{\gamma_7 - 1} \ (A_7 C_T p_7) \left[\left(\frac{P_7}{p_7} \right)^{\frac{\gamma_7 - 1}{\gamma_7}} - 1 \right] + A_7 C_T (p_7 - p_0) \tag{7}$$

When the jet velocity is supersonic, that is, the exhaust-nozzle pressure ratio P_7/p_0 is greater than 1.85, the static pressure at the outlet can be determined from the relation

$$p_7 = \frac{P_7}{\left(\frac{\gamma_7 + 1}{\gamma_7} \right)^{\frac{\gamma_7}{\gamma_7 - 1}}} \tag{8}$$

When the jet velocity is subsonic $(P_7/p_0) < 1.85$ and $p_7 = p_0$, then equation (7) becomes

$$F_{j,r} = \frac{2\gamma_7}{\gamma_7 - 1} \ (A_7 C_T p_0) \left[\left(\frac{P_7}{p_0} \right)^{\frac{\gamma_7 - 1}{\gamma_7}} - 1 \right] \tag{9}$$

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REFERENCE

1. Sanders, Newell D.: Performance Parameters for Jet-Propulsion Engines. NACA TN 1106, 1946.

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P_1/p_0

TABLE I - ENGINE

Run	Altitude (ft)	Ram pressure ratio P_1/p_0	Flight Mach number M_0	Tunnel static pressure p_0 (lb/sq ft abs.)	Compressor-inlet indicated temperature $T_{i,1}$ (°R)	Equivalent ambient temperature t_e (°R)	Engine speed N (rpm)	Jet thrust $F_{j,s}$ (lb)	Net thrust $F_{n,s}$ (lb)	Engine-inlet air flow $W_{a,1}$ (lb/sec)	Fuel flow W_f (lb/hr)	Specific fuel consumption $W_f/F_{n,s}$ (lb/(hr)(lb net thrust)).
1	5000	0.995	-----	1755	504	503	7955	----	----	83.15	5300	-----
2		1.022	0.176	1749	504	503	7386	----	----	82.21	4145	-----
3		1.022	.176	1757	594	593	6993	3679	3207	78.48	3405	1.062
4		1.022	.176	1764	506	505	6643	3101	2658	73.40	2900	1.091
5		1.025	.187	1750	506	503	5944	2166	1762	62.78	2110	1.198
6		1.027	.194	1755	505	502	5114	1296	967	49.39	1490	1.541
7		1.027	.194	1753	509	506	4091	640	413	33.94	1125	2.724
8		1.028	.197	1756	508	504	3147	556	192	24.10	855	4.453
9	6000	1.021	0.173	1693	528	527	7985	4768	4284	79.92	4890	1.141
10		1.018	.159	1686	509	508	7955	----	----	81.39	5095	-----
11		1.020	.169	1690	508	507	7692	4564	4096	80.89	4505	1.100
12		1.020	.169	1693	505	504	7386	4141	3683	79.37	3970	1.078
13		1.019	.164	1693	496	494	6993	3635	3209	76.58	3390	1.031
14		1.021	.173	1697	492	490	6643	3174	2742	73.36	2930	1.069
15		1.023	.180	1693	494	492	5944	2169	1791	62.11	2115	1.181
16		1.026	.190	1690	503	500	5114	1280	962	48.48	1475	1.533
17		1.028	.197	1686	501	498	4098	1653	422	34.17	1115	2.642
18		1.026	.190	1697	502	498	3147	332	178	23.51	855	4.803
19		1.028	.197	1690	494	490	2000	121	27	14.03	508	18.810
20	15,000	1.019	0.164	1188	464	463	7825	3762	3437	60.35	3918	1.139
21		1.020	.169	1190	470	469	7800	3700	3367	59.83	3775	1.121
22		1.018	.159	1191	473	473	7692	3489	3173	59.66	3465	1.092
23		1.021	.173	1190	473	472	7386	3142	2808	58.36	2970	1.058
24		1.020	.169	1193	472	470	6993	2717	2402	56.47	2500	1.041
25		1.023	.180	1190	474	472	6643	2366	2045	53.70	2150	1.051
26		1.024	.183	1190	473	471	5944	1633	1355	45.60	1570	1.159
27		1.029	.201	1189	476	472	5114	969	725	36.45	1080	1.490
28		1.027	.194	1191	476	472	4091	473	313	24.75	820	2.620
29		1.031	.207	1181	477	473	3147	267	160	15.33	667	4.169
30		1.028	.197	1204	473	469	1750	48	-18	10.22	330	-----
31		1.191	.509	1191	498	476	7955	4506	3357	68.30	4310	1.284
32		1.155	.500	1197	499	477	7692	4078	2956	67.54	3680	1.245
33		1.192	.509	1195	500	477	7386	3655	2545	65.76	3155	1.240
34		1.194	.510	1190	493	470	6993	3198	2134	63.19	2705	1.321
35		1.191	.509	1188	493	470	6643	2749	1749	59.82	2275	1.301
36	25,000	1.022	0.176	778	460	459	7875	2512	2283	39.85	2690	1.178
37		1.023	.180	777	459	458	7692	2343	2113	39.22	2460	1.164
38		1.021	173	778	459	458	7386	2118	1901	38.49	2125	1.118
39		1.022	.176	778	460	458	6993	1812	1599	37.10	1764	1.103
40		1.022	.176	781	457	455	6643	1576	1369	36.12	1536	1.122
41		1.022	.176	779	454	452	5944	1119	940	31.35	1113	1.184
42		1.027	.194	781	455	452	5114	654	495	25.21	796	1.608
43		1.026	.190	778	454	451	4091	299	192	17.18	638	3.323
44		1.031	.207	781	458	454	3147	147	70	11.32	521	7.443
45		1.028	.197	781	464	460	2046	39	-15	8.27	302	-----
46		1.188	.500	786	457	437	7900	3239	2483	47.19	3225	1.299
47		1.196	.511	781	454	433	7692	3016	2256	46.83	2845	1.261
48		1.192	.509	781	455	435	7386	2719	1980	45.81	2440	1.232
49		1.190	.508	779	455	434	6993	2324	1614	44.30	1985	1.230
50		1.192	.509	778	454	433	6643	2024	1326	43.36	1656	1.249
51		1.196	.511	781	455	433	5944	1375	767	37.44	1113	1.451
52		1.201	.521	781	457	434	5114	791	300	29.82	676	2.253
53		1.206	.526	778	457	433	4091	360	15	20.75	482	32.130
54		1.399	.711	777	468	427	7900	3880	2660	54.57	3470	1.306
55		1.399	.711	777	468	427	7692	3659	2454	53.94	3175	1.294
56		1.399	.711	781	468	427	7386	3380	2168	53.35	2720	1.255
57		1.395	.709	778	467	425	6993	2900	1771	50.86	2220	1.254
58		1.409	.719	777	470	427	6643	2522	1419	48.83	1855	1.307
59		1.403	.716	781	471	428	5944	1674	723	42.30	1105	1.528
60		1.409	.719	778	468	425	5914	1646	689	42.43	1094	1.588
61		1.408	.719	781	472	428	4600	604	-55	29.18	420	-----
62	35,000	1.018	0.159	496	441	441	7750	1586	1456	25.42	1786	1.227
63		1.018	.159	494	441	441	7692	1588	1459	25.29	1741	1.193
64		1.022	.176	496	441	440	7386	1403	1261	25.22	1429	1.133
65		1.018	.159	495	441	440	6993	1228	1106	23.87	1184	1.071
66		1.018	.159	499	445	444	6643	1061	942	23.29	1003	1.065
67	45,000	1.023	0.180	304	437	436	7525	958	869	15.55	1068	1.229
68		1.030	.205	303	443	441	7550	956	852	15.75	1098	1.289
69		1.030	.205	303	444	441	7500	937	835	15.48	1048	1.255
70		1.020	.169	303	438	437	7383	877	797	14.86	979	1.228
71		1.033	.216	303	442	439	7386	894	787	15.57	962	1.222
72		1.026	.190	303	441	439	6993	750	658	14.94	796	1.210
73		1.023	.183	306	440	438	6643	682	600	14.35	701	1.168
74		1.023	.180	299	440	438	6500	627	548	13.76	680	1.241
75		1.029	.200	308	440	437	6294	----	----	14.10	624	-----
76		1.026	.190	310	440	438	5944	462	383	12.96	557	1.454
77		1.033	.218	306	440	437	5455	346	266	11.96	490	1.842
78		1.029	.200	306	440	436	5114	275	208	10.40	454	2.183
79		1.032	.211	310	440	436	4545	170	119	7.50	431	3.622
80		1.023	.183	303	440	437	3977	123	85	6.55	454	5.341
81	55,000	1.027	0.192	186	438	434	7386	545	485	9.64	682	1.406
82		1.038	.228	183	438	434	7343	539	471	9.32	640	1.359
83		1.038	.228	185	437	433	6993	481	413	9.34	560	1.356
84		1.031	.210	191	437	434	6643	428	369	8.85	527	1.428
85		1.042	.242	189	438	434	6250	363	295	8.81	486	1.647

[a]Calculated values.

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PERFORMANCE DATA

Fuel-air ratio f/a	Exhaust-gas total temperature, T_7 (°R)	Turbine-outlet total pressure P_6 (lb/sq ft abs.)	Corrected engine speed $N/\sqrt{\theta}$ (rpm)	Corrected net thrust $F_{n,s}/\delta$ (lb)	Corrected engine-inlet air flow $W_{a,1}\sqrt{\theta}/\delta$ (lb/sec)	Corrected fuel flow $W_f/\delta\sqrt{\theta}$ (lb/hr)	Corrected specific fuel consumption $W_f/F_{n,s}\sqrt{\theta}$ (lb/(hr) (lb net thrust))	Corrected fuel-air ratio (f/a)/θ	Corrected exhaust-gas total temperature T_7/θ (°R)	Engine total-pressure ratio P_7/P_1	Engine total-temperature ratio T_7/T_1	Run
0.0177	----	3559	----	----	-----	----	-----	------	----	1.962	-----	1
.0140	----	3262	7504	----	97.91	5096	-----	0.0145	----	1.761	-----	2
.0121	----	3006	7105	3861	93.00	4165	1.079	.0124	----	1.619	-----	3
.0110	1299	2809	6736	3190	86.86	3529	1.106	.0113	1336	1.508	2.557	4
.0093	1165	2428	6039	2130	74.71	2591	1.217	.0096	1202	1.317	2.298	5
.0084	1092	2153	5201	1166	58.57	1826	1.567	.0087	1129	1.178	2.158	6
.0092	1141	1953	4144	498	40.44	1375	2.759	.0094	1171	1.081	2.237	7
.0099	1175	1867	3194	231	28.61	1045	4.520	.0102	1210	1.035	2.313	8
0.0170	----	3391	7895	5356	100.70	6066	1.133	0.0167	----	1.887	-----	9
.0174	----	3462	8043	----	101.00	6464	-----	.0178	----	1.938	-----	10
.0155	1633	3273	7784	5128	100.10	5708	1.113	.0158	1671	1.848	3.203	11
.0139	1513	3145	7497	4604	97.75	5036	1.094	.0143	1558	1.760	2.984	12
.0123	1395	2942	7168	4011	93.39	4343	1.083	.0129	1466	1.651	2.807	13
.0111	1303	2759	6836	3419	88.90	3759	1.100	.0117	1380	1.546	2.643	14
.0095	1158	2387	6104	2239	75.60	2715	1.213	.0100	1220	1.346	2.339	15
.0085	1094	2090	5211	1204	59.57	1681	1.562	.0088	1135	1.187	2.171	16
.0091	1122	1884	4184	530	42.00	1428	2.698	.0094	1169	1.084	2.235	17
.0101	1159	1798	3213	222	28.71	1088	4.904	.0105	1208	1.032	2.309	18
.0101	1070	1741	2058	34	17.07	654	19.360	.0101	1133	1.002	2.166	19
0.0180	1722	2581	8287	6132	101.67	7396	1.206	0.0202	1930	2.038	3.695	20
.0175	----	2535	8206	5987	101.12	7061	1.179	.0194	----	1.988	-----	21
.0161	1623	2448	8054	5638	101.26	6446	1.143	.0177	1781	1.939	3.417	22
.0141	1482	2307	7748	4993	98.92	5540	1.110	.0156	1630	1.826	3.120	23
.0123	1359	2132	7350	4261	95.32	4661	1.094	.0136	1500	1.688	2.873	24
.0111	1279	2001	6969	3636	91.02	4010	1.103	.0122	1406	1.588	2.693	25
.0096	1129	1744	6241	2409	77.22	2931	1.217	.0105	1246	1.390	2.382	26
.0082	1044	1509	5365	1291	61.85	2018	1.563	.0091	1148	1.212	2.193	27
.0092	1079	1349	4291	556	41.93	1528	2.748	.0101	1187	1.095	2.287	28
.0121	1108	1268	3295	287	26.24	1251	4.365	.0133	1215	1.039	2.323	29
.0090	971	1229	1641	----	17.07	610	-----	.0099	1075	.992	2.053	30
.0175	1732	2889	8305	5965	116.30	7996	1.340	.0191	1888	1.947	3.464	31
.0151	1591	2716	8023	5226	114.50	6786	1.298	.0165	1731	1.846	3.176	32
.0133	1470	2540	7704	4507	111.70	5828	1.293	.0145	1600	1.723	2.928	33
.0119	1355	2360	7350	3794	106.90	5054	1.332	.0131	1495	1.605	2.743	34
.0106	1256	2165	6982	3115	101.40	4259	1.367	.0117	1386	1.482	2.543	35
0.0188	1727	1707	8371	6210	102.00	7778	1.253	0.0212	1955	2.057	3.738	36
.0174	1643	1646	8184	5754	100.40	7128	1.239	.0197	1860	1.991	3.564	37
.0153	1520	1562	7859	5171	98.40	6150	1.189	.0174	1722	1.901	3.297	38
.0132	1391	1444	7441	4349	94.84	5105	1.174	.0150	1576	1.762	3.017	39
.0118	1300	1352	7095	3709	91.62	4444	1.198	.0135	1484	1.644	2.838	40
.0099	1118	1171	6366	2553	79.50	3238	1.268	.0113	1284	1.431	2.457	41
.0088	1011	1003	5477	1341	63.77	2309	1.722	.0101	1160	1.227	2.222	42
.0103	1050	890	4590	522	43.55	1862	3.565	.0119	1209	1.107	2.313	43
.0128	1069	844	3364	190	28.69	1508	7.956	.0146	1222	1.047	2.334	44
.0101	1077	804	2173	----	21.10	869	-----	.0114	1215	1.001	2.321	45
.0190	1740	2026	8611	6684	116.50	9463	1.416	.0226	2068	2.078	3.791	46
.0169	1630	1928	8423	6112	115.90	8439	1.381	.0202	1955	1.976	3.575	47
.0148	1488	1800	8066	5364	113.60	7218	1.346	.0177	1775	1.873	3.256	48
.0124	1341	1647	7650	4384	110.00	5898	1.345	.0143	1604	1.725	2.941	49
.0106	1240	1511	7274	3607	107.70	4932	1.368	.0127	1486	1.578	2.725	50
.0083	1020	1262	6509	2078	92.63	3301	1.589	.0099	1223	1.316	2.237	51
.0063	864	1040	5595	813	73.84	2003	2.465	.0075	1034	1.093	1.891	52
.0065	809	902	4480	41	51.54	1436	35.190	.0077	970	.953	1.770	53
.0177	1707	2291	8706	7243	134.84	10412	1.438	.0215	2074	2.011	3.632	54
.0163	1623	2198	8477	6682	133.28	9527	1.426	.0199	1972	1.936	3.453	55
.0142	1484	2063	8139	5873	131.15	8120	1.383	.0172	1802	1.812	3.157	56
.0121	1337	1863	7727	4817	125.19	6672	1.385	.0148	1676	1.663	2.857	57
.0106	1235	1721	7321	3864	120.66	5566	1.441	.0128	1500	1.514	2.622	58
.0073	979	1379	6544	1959	104.08	3295	1.683	.0088	1188	1.215	2.074	59
.0072	965	1383	6535	1874	104.44	3287	1.755	.0087	1178	1.214	2.058	60
.0040	712	983	5065	----	71.80	1252	-----	.0048	864	.878	1.508	61
0.0195	1757	1116	8409	6211	99.95	8267	1.331	0.0230	2070	2.105	3.966	62
.0191	1730	1095	8346	6249	99.83	8091	1.295	.0225	2036	2.087	3.905	63
.0157	1529	1016	8021	5379	99.07	6620	1.231	.0186	1805	1.931	3.451	64
.0138	1396	931	7594	4728	93.96	5497	1.163	.0163	1647	1.784	3.158	65
.0120	1293	888	7181	3994	91.35	4597	1.151	.0140	1511	1.683	2.899	66
0.0191	1725	659	8210	6049	99.21	8111	1.341	0.0227	2052	2.026	3.929	67
.0193	1734	----	8192	5933	101.09	8296	1.398	.0228	2041	2.035	3.897	68
.0188	1710	----	8138	5815	99.36	7918	1.362	.0221	2012	[a]2.010	3.843	69
.0183	1635	631	8051	5565	95.20	7452	1.339	.0217	1943	1.961	3.724	70
.0171	1632	----	8029	5481	99.75	7282	1.329	.0203	1932	[a]1.955	3.684	71
.0148	1463	----	7601	4582	95.71	6025	1.315	.0175	1730	[a]1.791	3.310	72
.0135	1346	----	7234	4149	91.12	5278	1.272	.0161	1595	[a]1.684	3.052	73
.0137	1318	512	7079	3878	89.42	5241	1.354	.0163	1561	1.601	2.989	74
.0123	----	----	6860	----	88.87	4673	-----	.0146	----	-----	-----	75
.0119	1159	----	6473	2614	81.24	4140	1.584	.0141	1374	[a]1.431	2.628	76
.0116	1091	----	5946	1839	74.16	3693	2.008	.0138	1285	[a]1.307	2.474	77
.0121	1049	----	5579	1438	65.92	3425	2.381	.0144	1248	[a]1.213	2.384	78
.0160	1086	----	4959	812	46.92	3210	3.951	.0190	1294	[a]1.069	2.468	79
.0193	1159	----	4335	592	41.85	3447	5.822	.0229	1377	[a]1.135	2.634	80
0.0196	1739	----	8080	5519	100.28	8491	1.538	0.0235	2080	[a]2.016	3.979	80
.0191	1677	----	8033	5445	98.48	8093	1.487	.0228	2008	[a]1.963	3.820	81
.0167	1532	----	7657	4725	97.58	7015	1.485	.0200	1836	[a]1.807	3.498	82
.0166	1413	----	7267	4089	89.63	6388	1.562	.0198	1690	[a]1.721	3.226	83
.0153	1293	----	6838	3304	90.19	5955	1.802	.0228	1546	[a]1.553	2.945	84

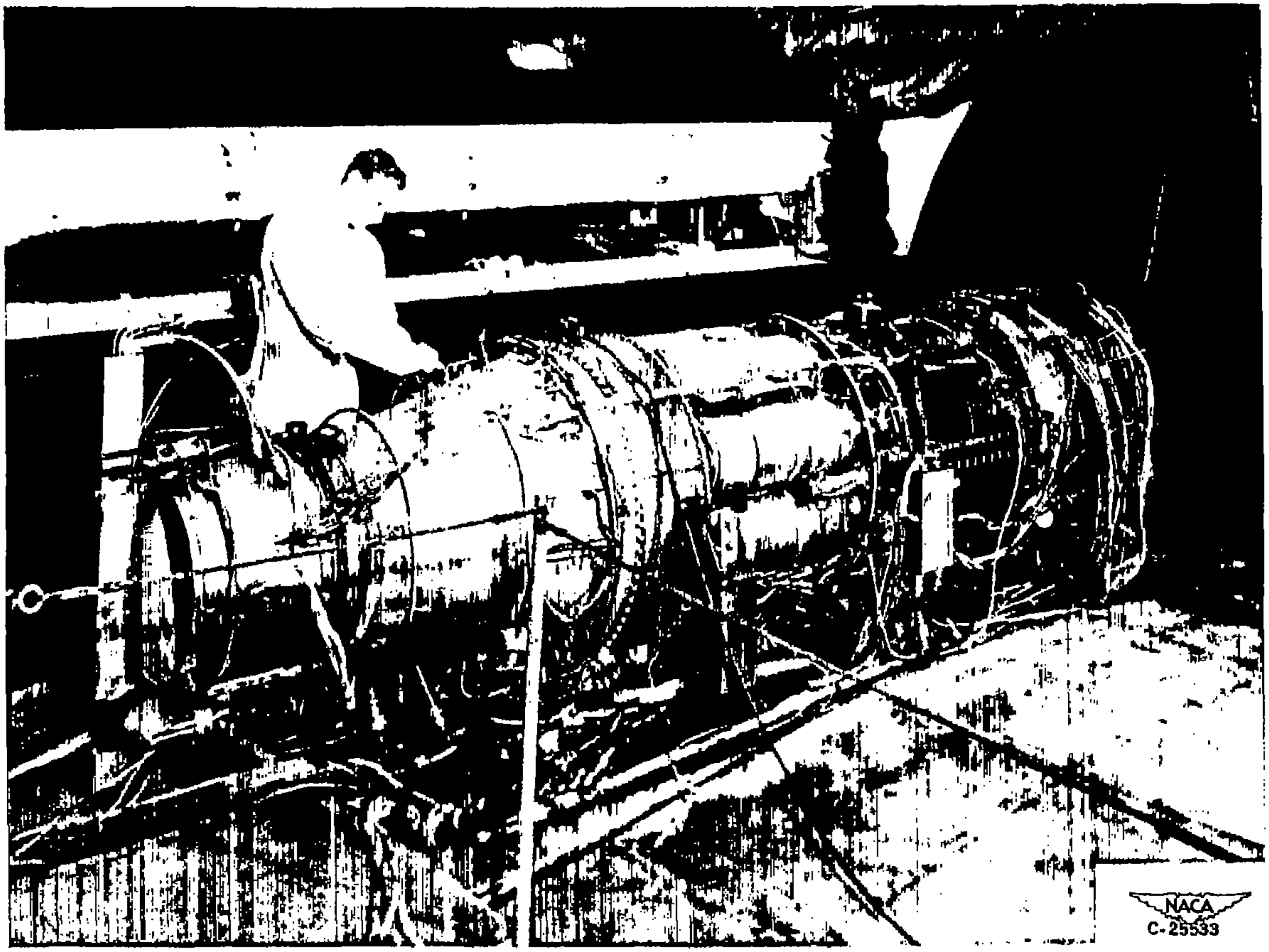
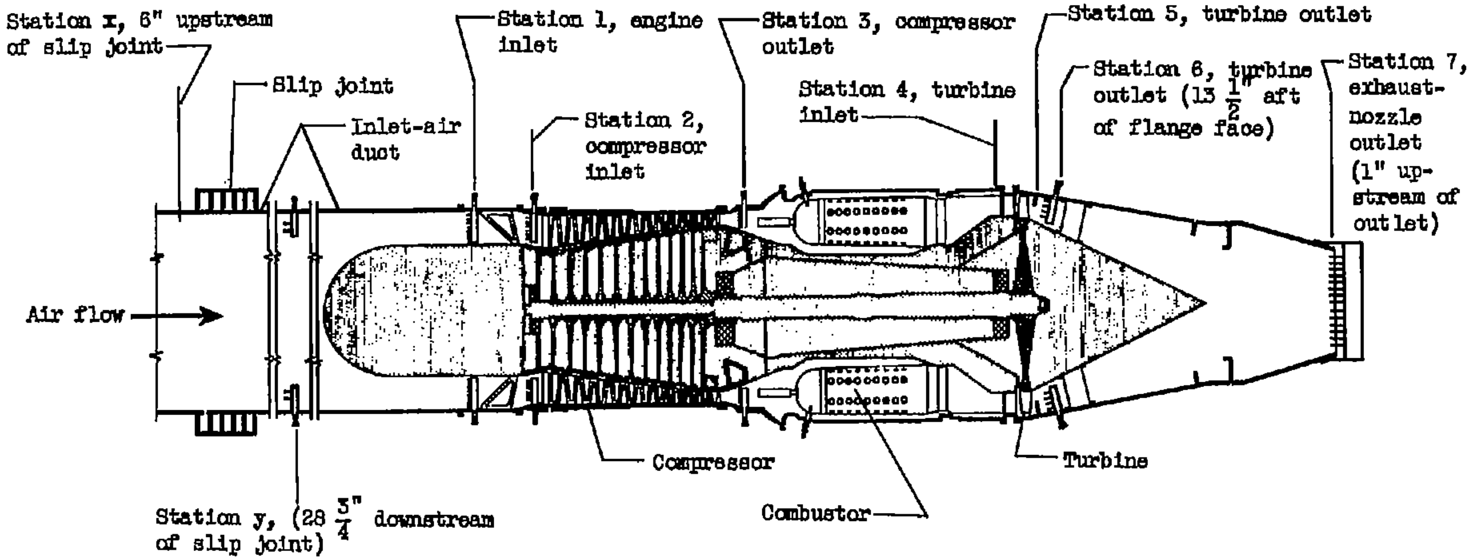


Figure 1. - The J47D (RX1-1) turbojet engine installed in test section of altitude wind tunnel.



Station	Total-pressure tubes	Static-pressure tubes	Wall static orifices	Thermo-couples
x	0	0	4	0
y	1	2	4	2
1	32	8	5	4
2	6	0	2	0
3	20	0	4	6
4	5	0	0	0
5	5	0	0	0
6	30	0	4	24
7	22	3	4	10

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Figure 2. - Cross section of engine showing location of instrumentation.

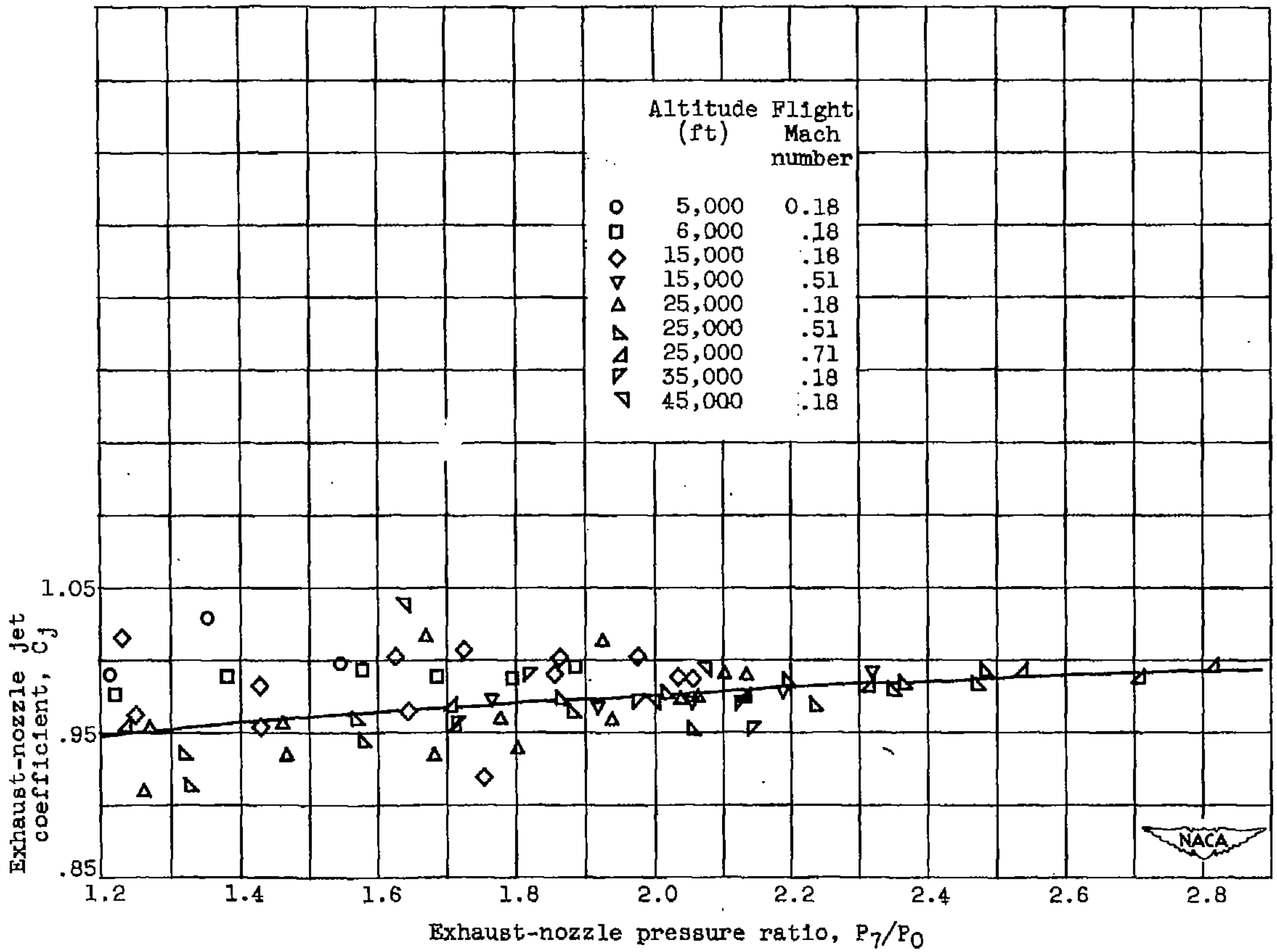
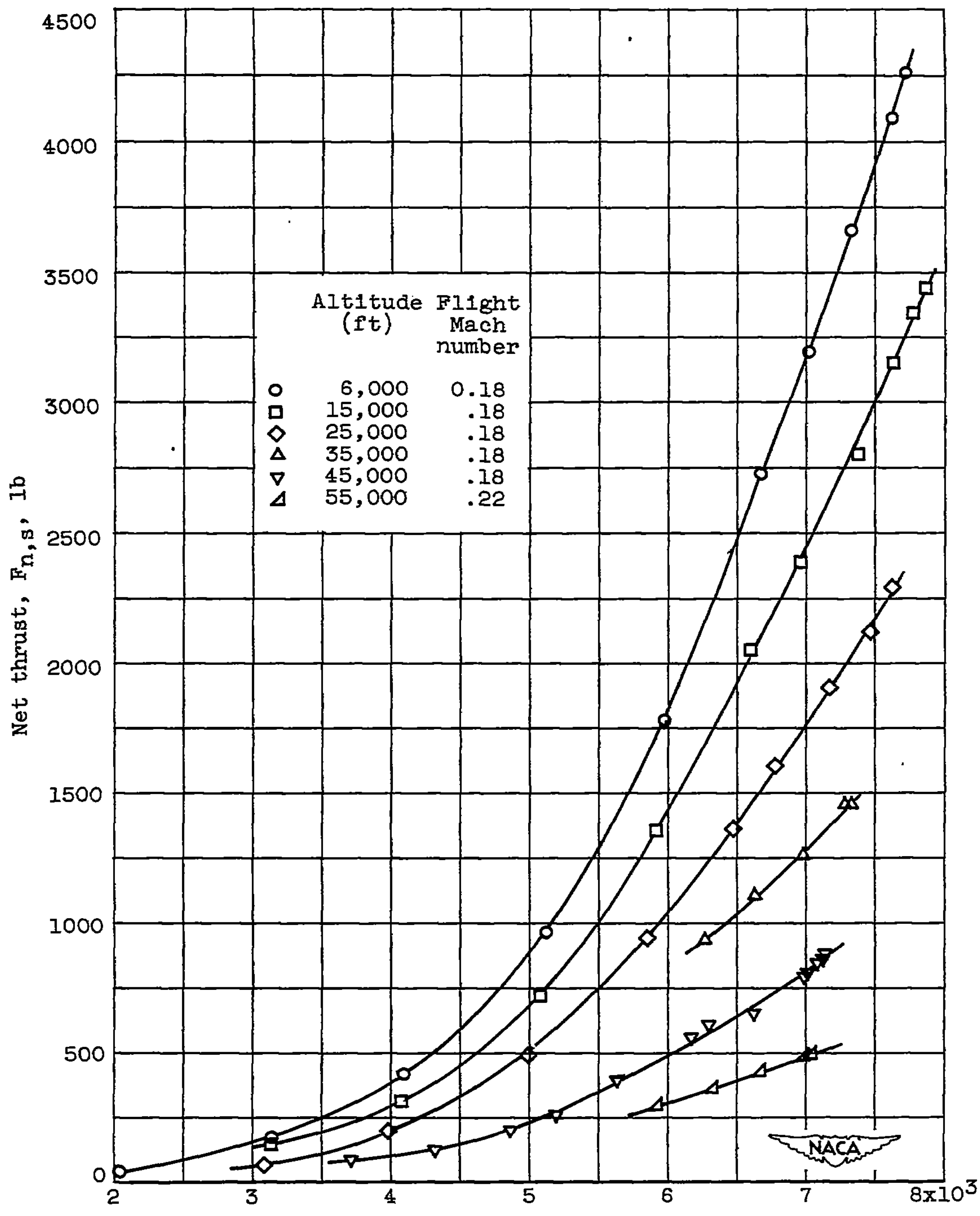


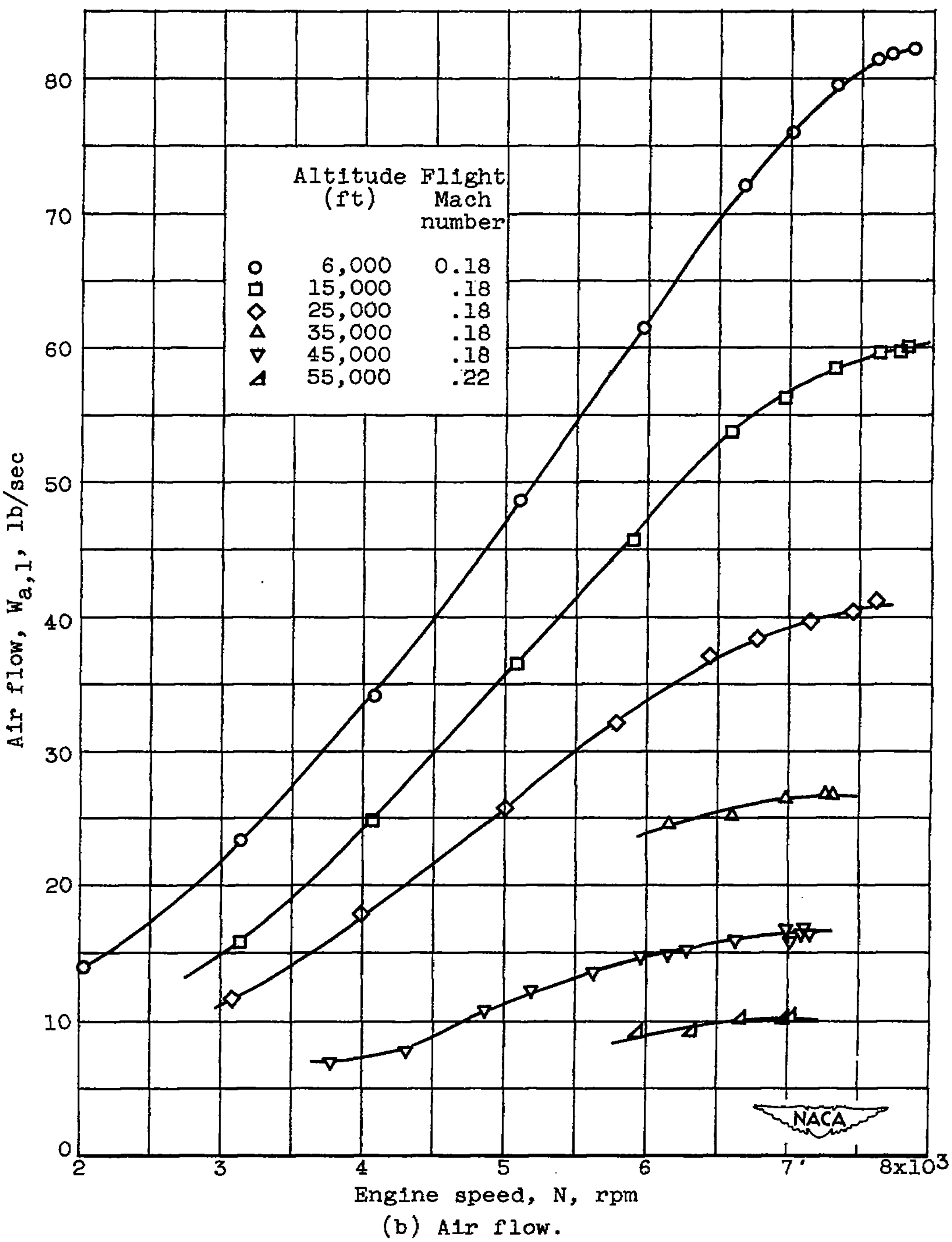
Figure 3. - Variation of exhaust-nozzle jet coefficient with exhaust-nozzle pressure ratio.



(a) Net thrust.

Figure 4. - Effect of altitude on variation of engine performance with engine speed.

2108

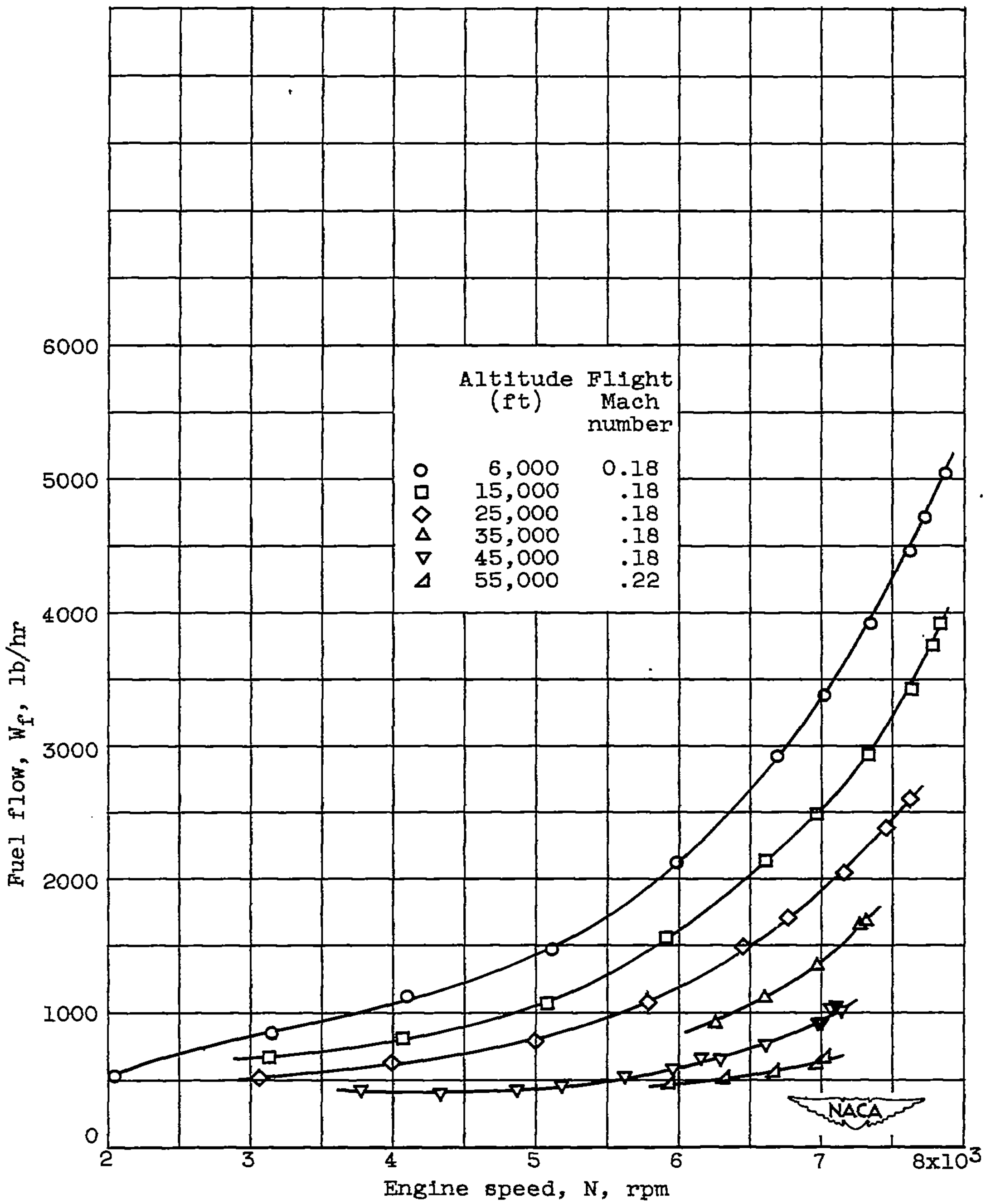


(b) Air flow.

Figure 4. - Continued. Effect of altitude on variation of engine performance with engine speed.

[redacted]

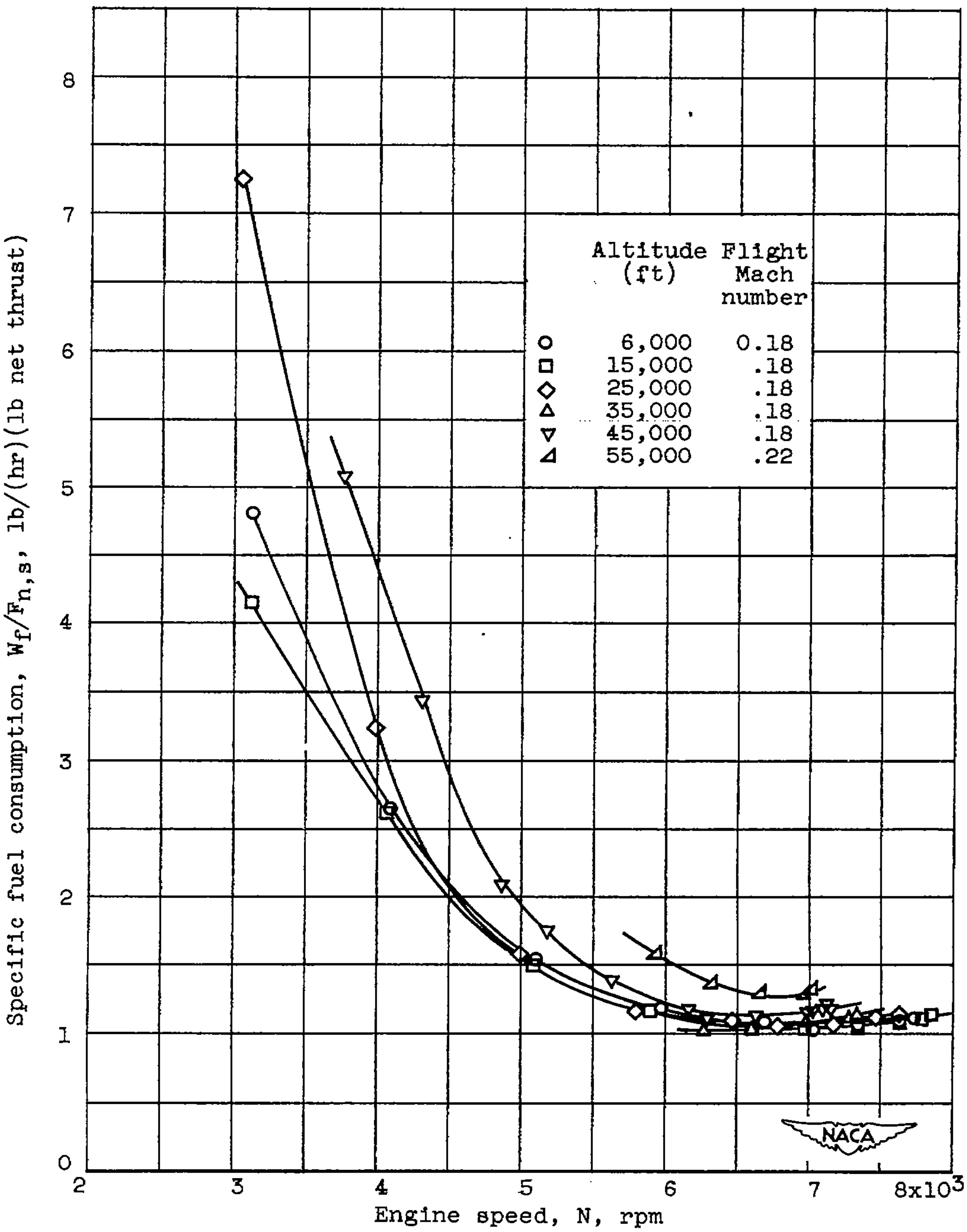
2108



(c) Fuel flow.

Figure 4. - Continued. Effect of altitude on variation of engine performance with engine speed.

[redacted]

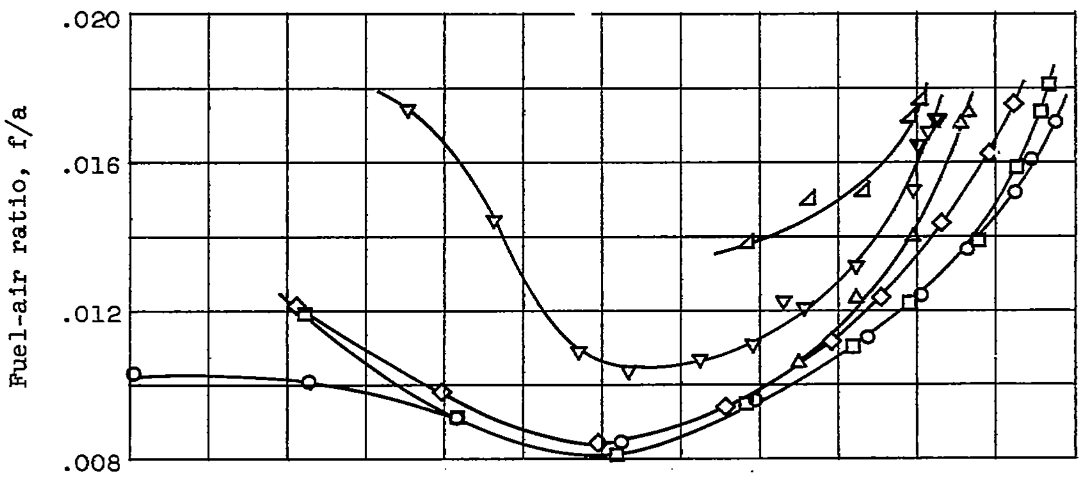


(d) Specific fuel consumption.

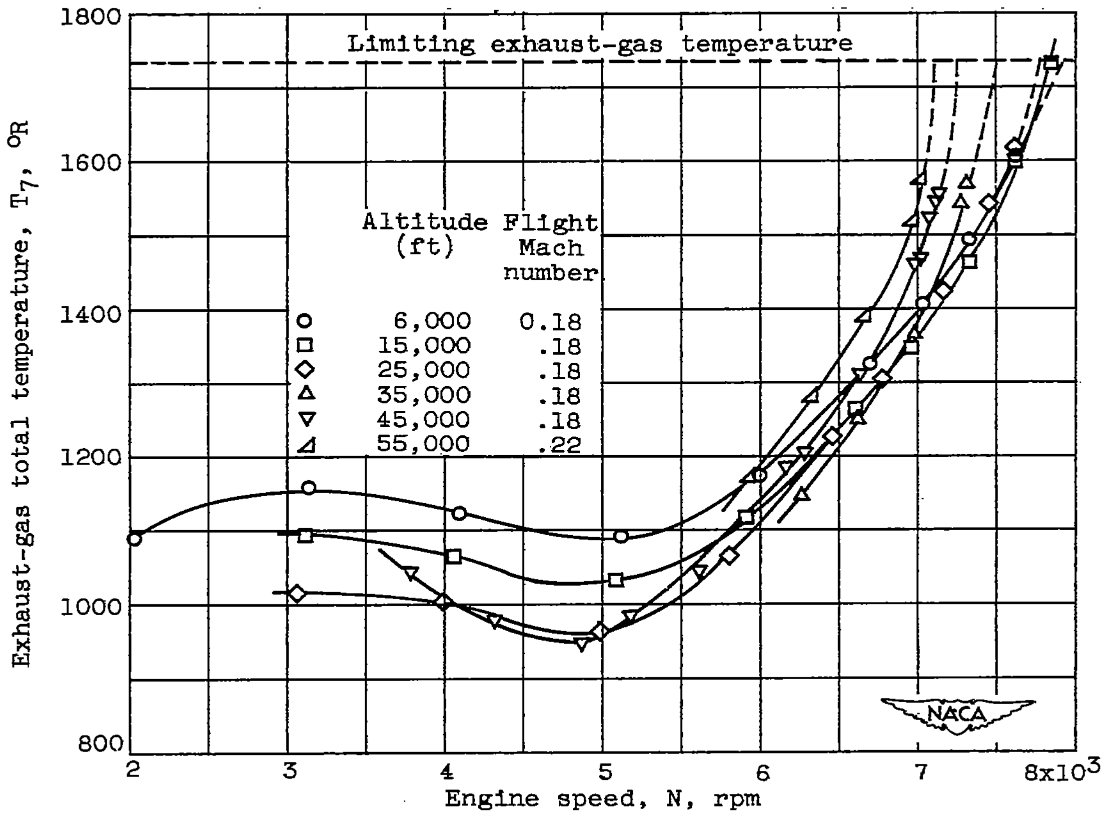
Figure 4. - Continued. Effect of altitude on variation of engine performance with engine speed.

[redacted]

2108



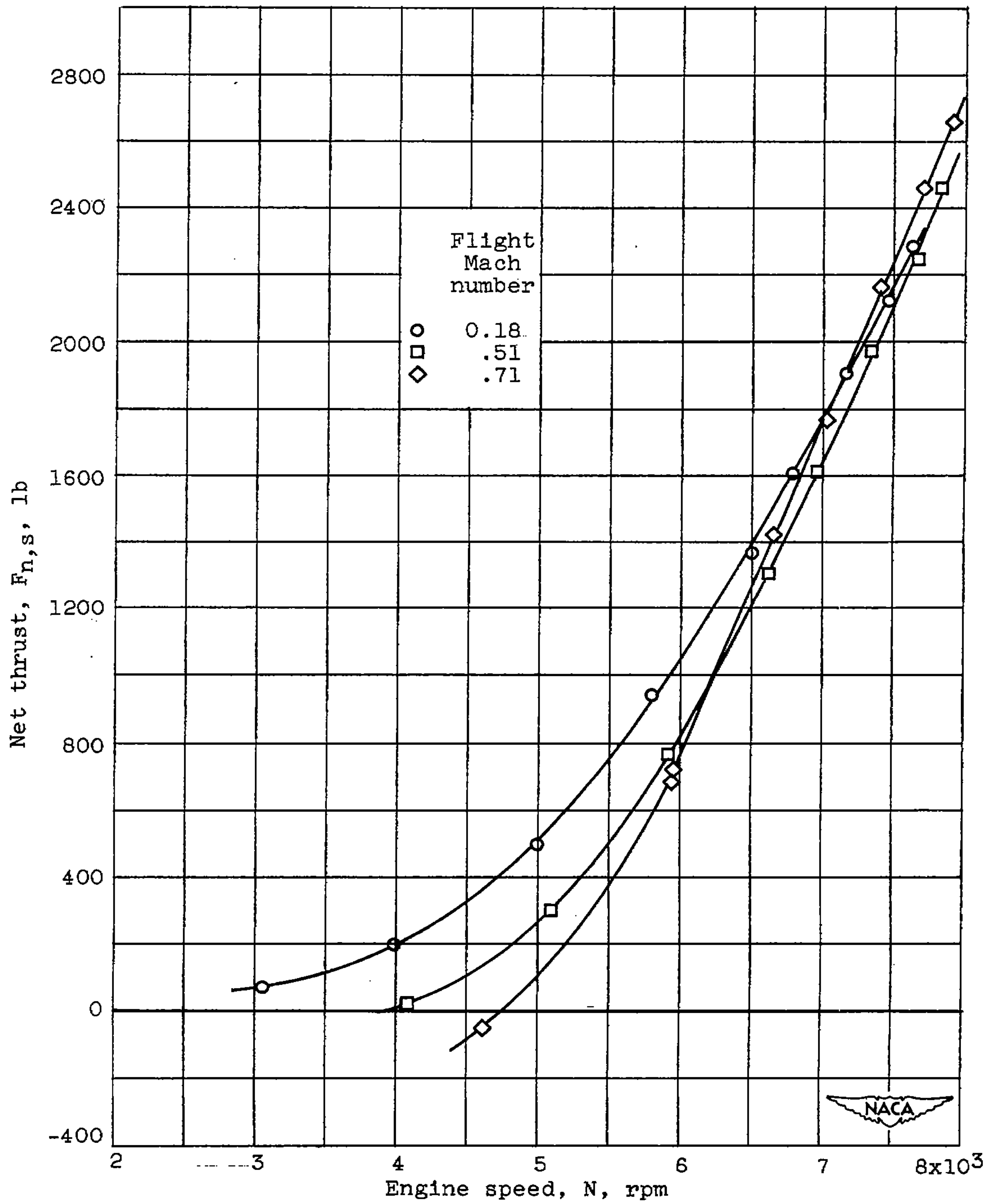
(e) Fuel-air ratio.



(f) Exhaust-gas total temperature.

Figure 4. - Concluded. Effect of altitude on variation of engine performance with engine speed.

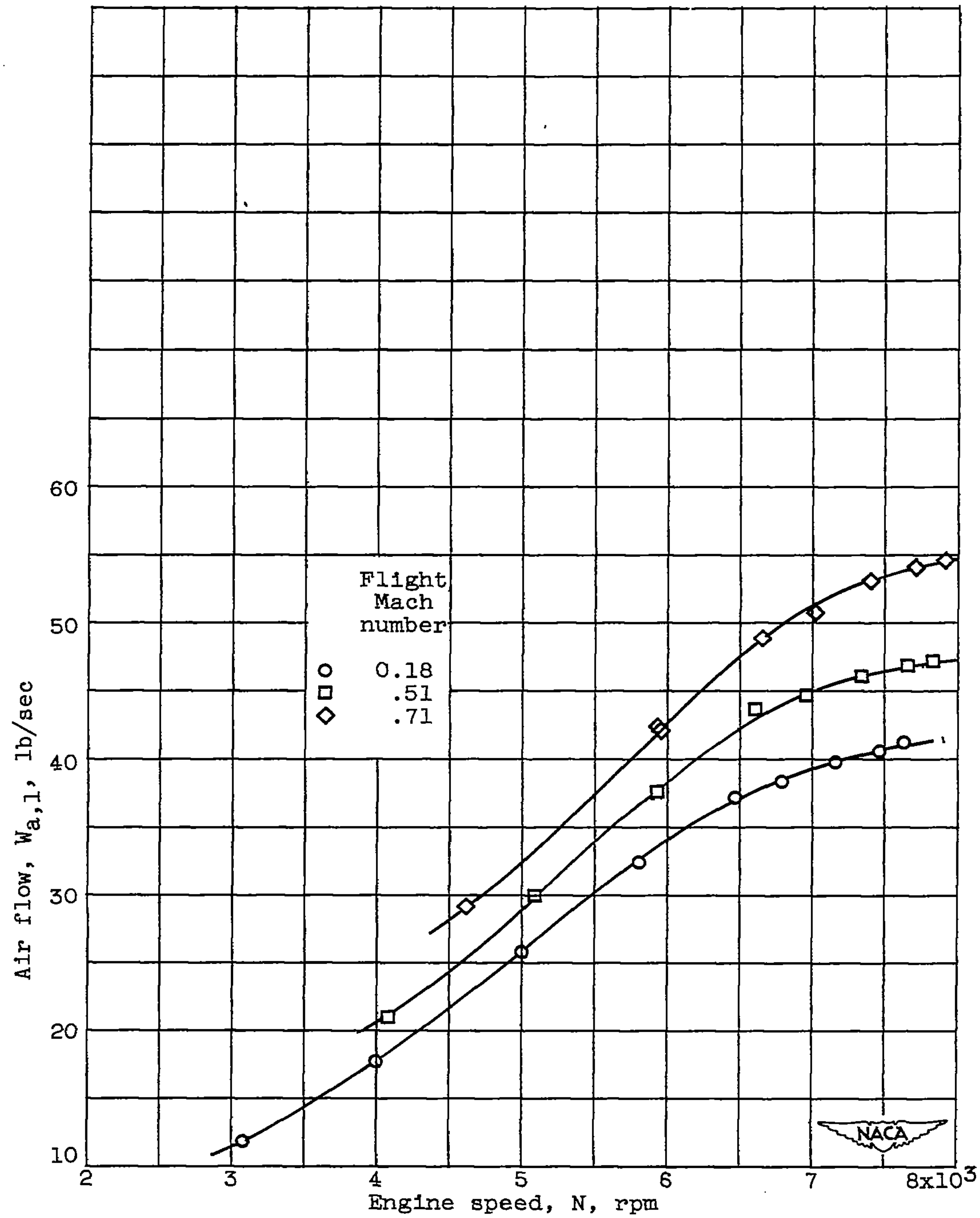
2108



(a) Net thrust.

Figure 5. - Effect of flight Mach number on variation of engine performance with engine speed at altitude of 25,000 feet.

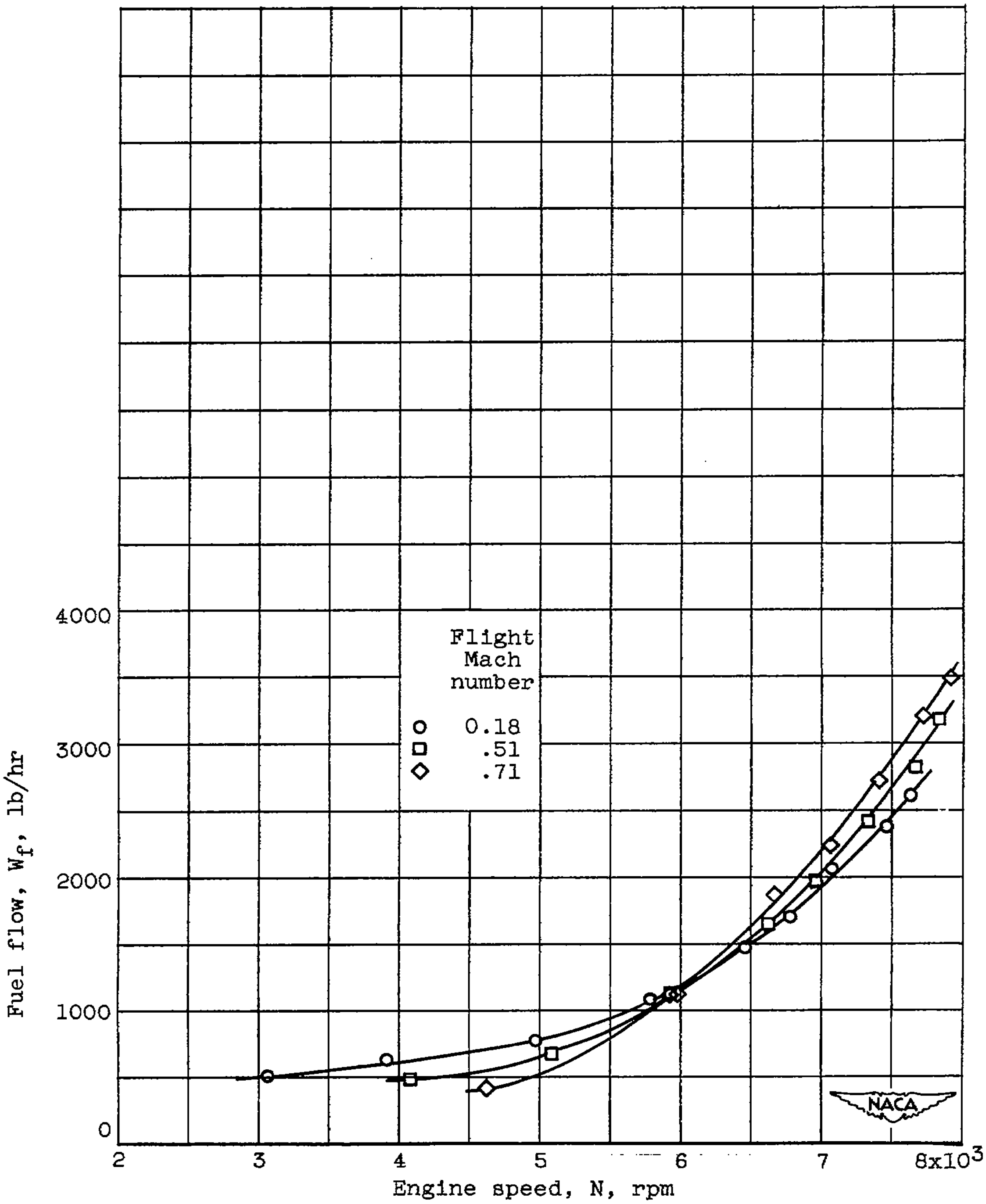
2108



(b) Air flow.

Figure 5. - Continued. Effect of flight Mach number on variation of engine performance with engine speed at altitude of 25,000 feet.

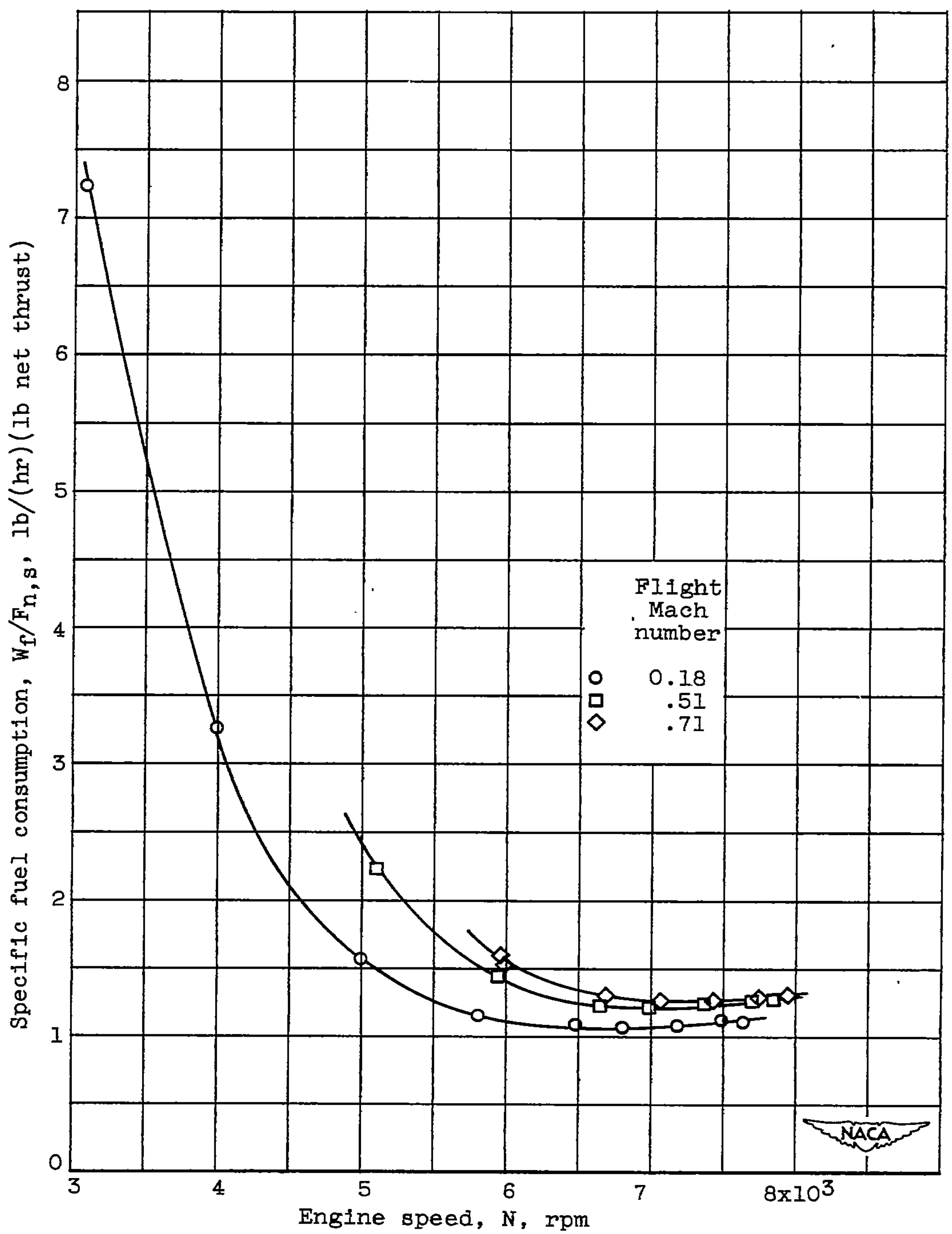
[redacted]



(c) Fuel flow.

Figure 5. - Continued. Effect of flight Mach number on variation of engine performance with engine speed at altitude of 25,000 feet.

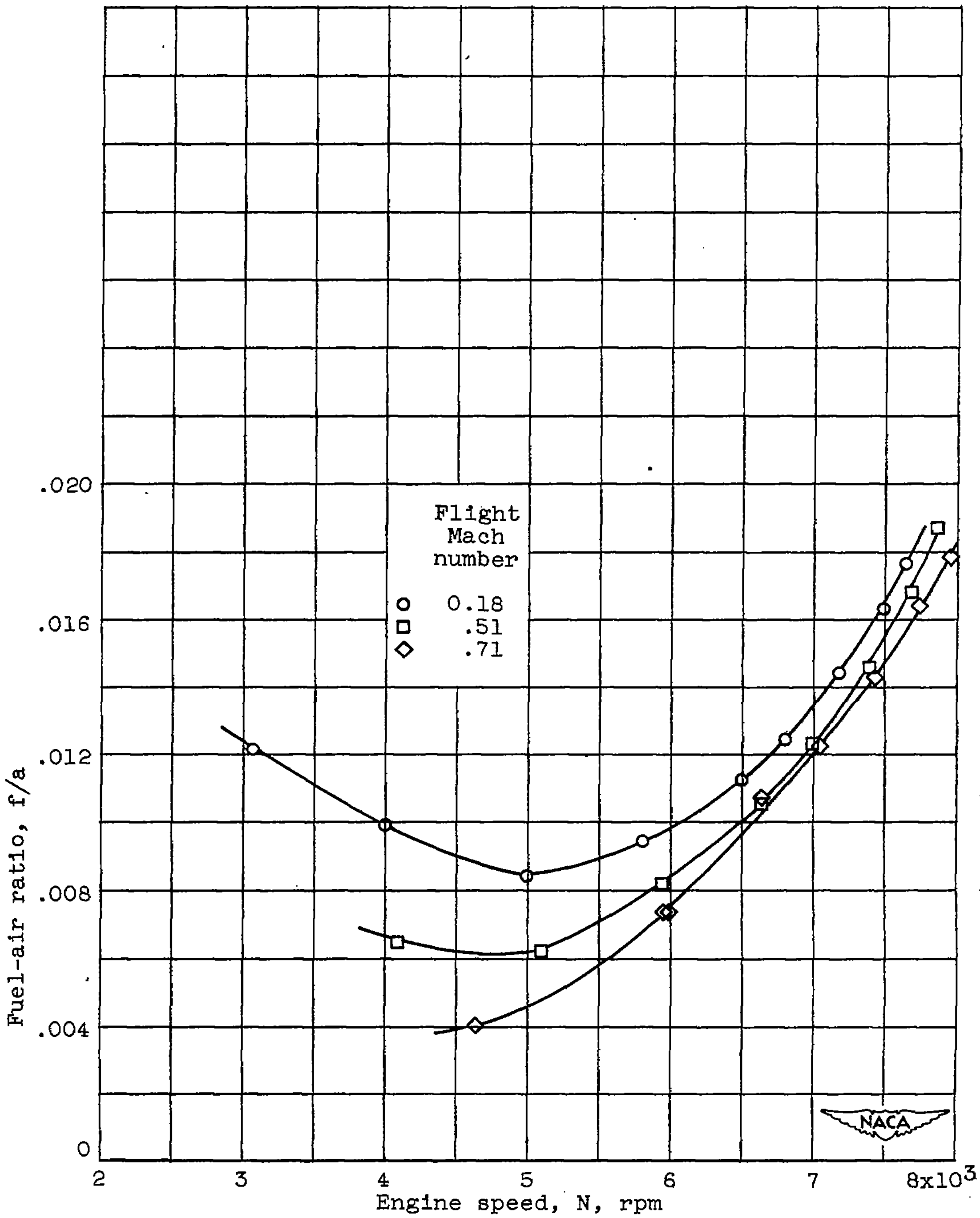
2108



(d) Specific fuel consumption.

Figure 5. - Continued. Effect of flight Mach number on variation of engine performance with engine speed at altitude of 25,000 feet

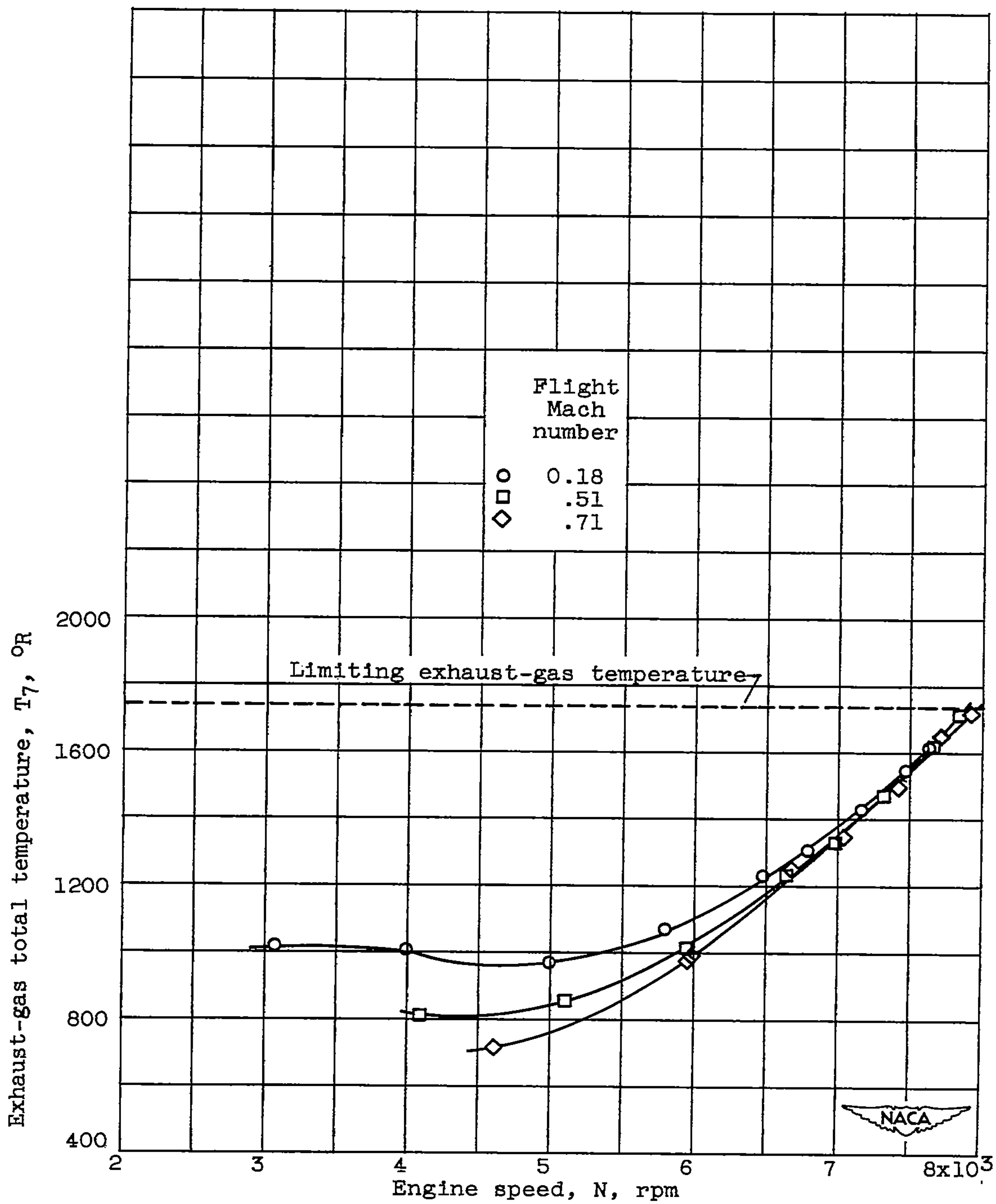
2108



(e) Fuel-air ratio.

Figure 5. - Continued. Effect of flight Mach number on variation of engine performance with engine speed at altitude of 25,000 feet.

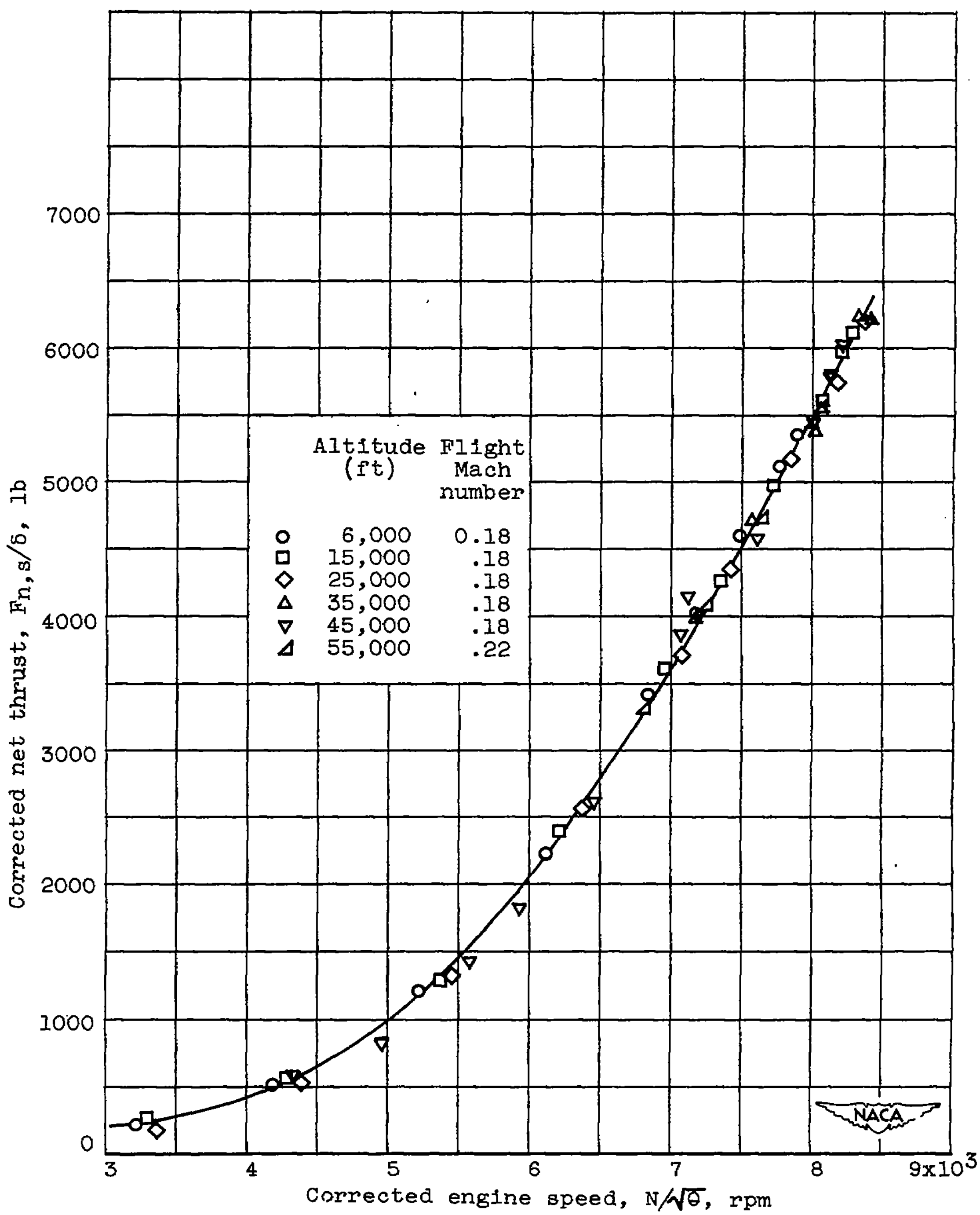
2108



(f) Exhaust-gas total temperature.

Figure 5. - Concluded. Effect of flight Mach number on variation of engine performance with engine speed at altitude of 25,000 feet.

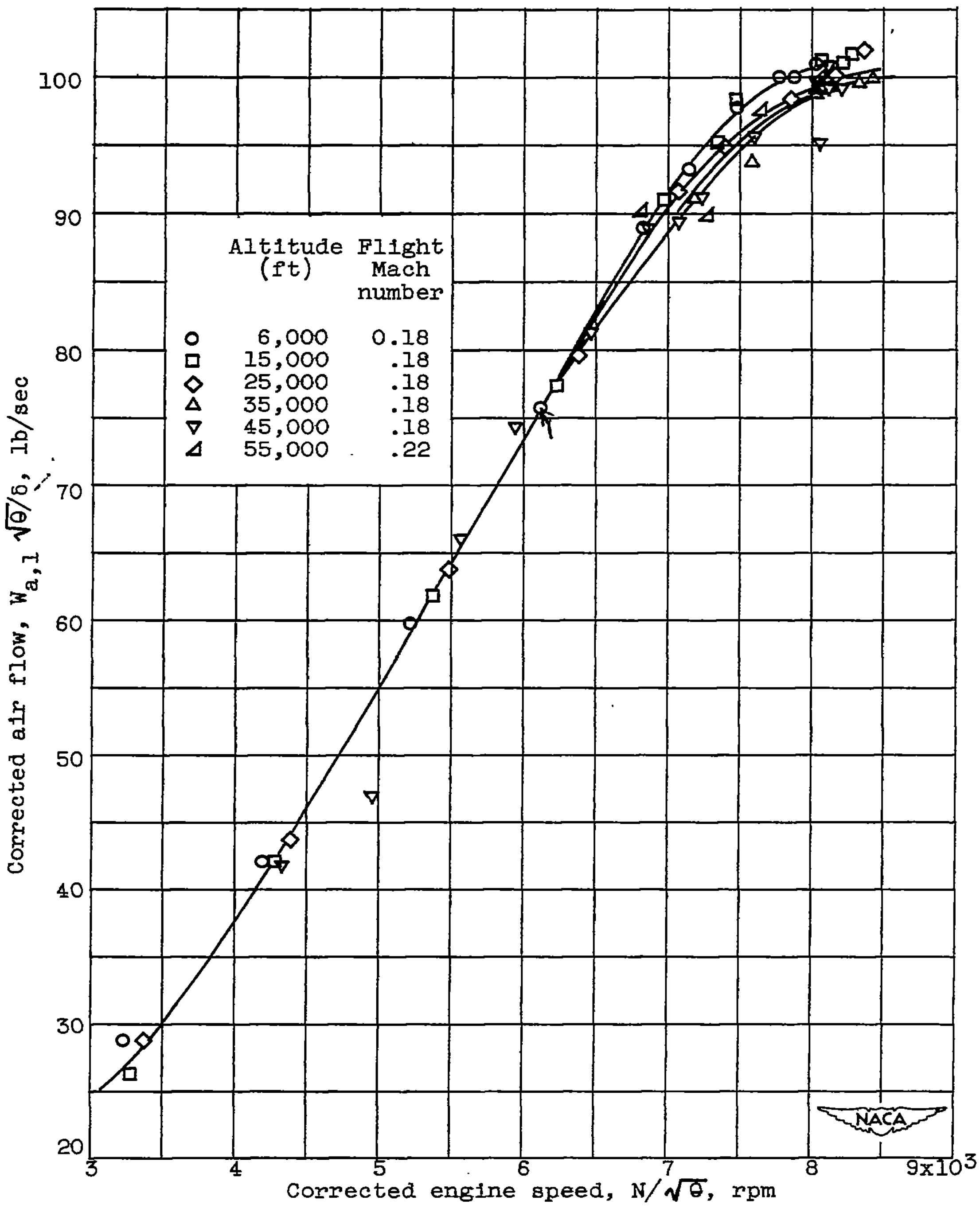
[redacted]



(a) Net thrust.

Figure 6. - Effect of altitude on variation of corrected engine performance with corrected engine speed.

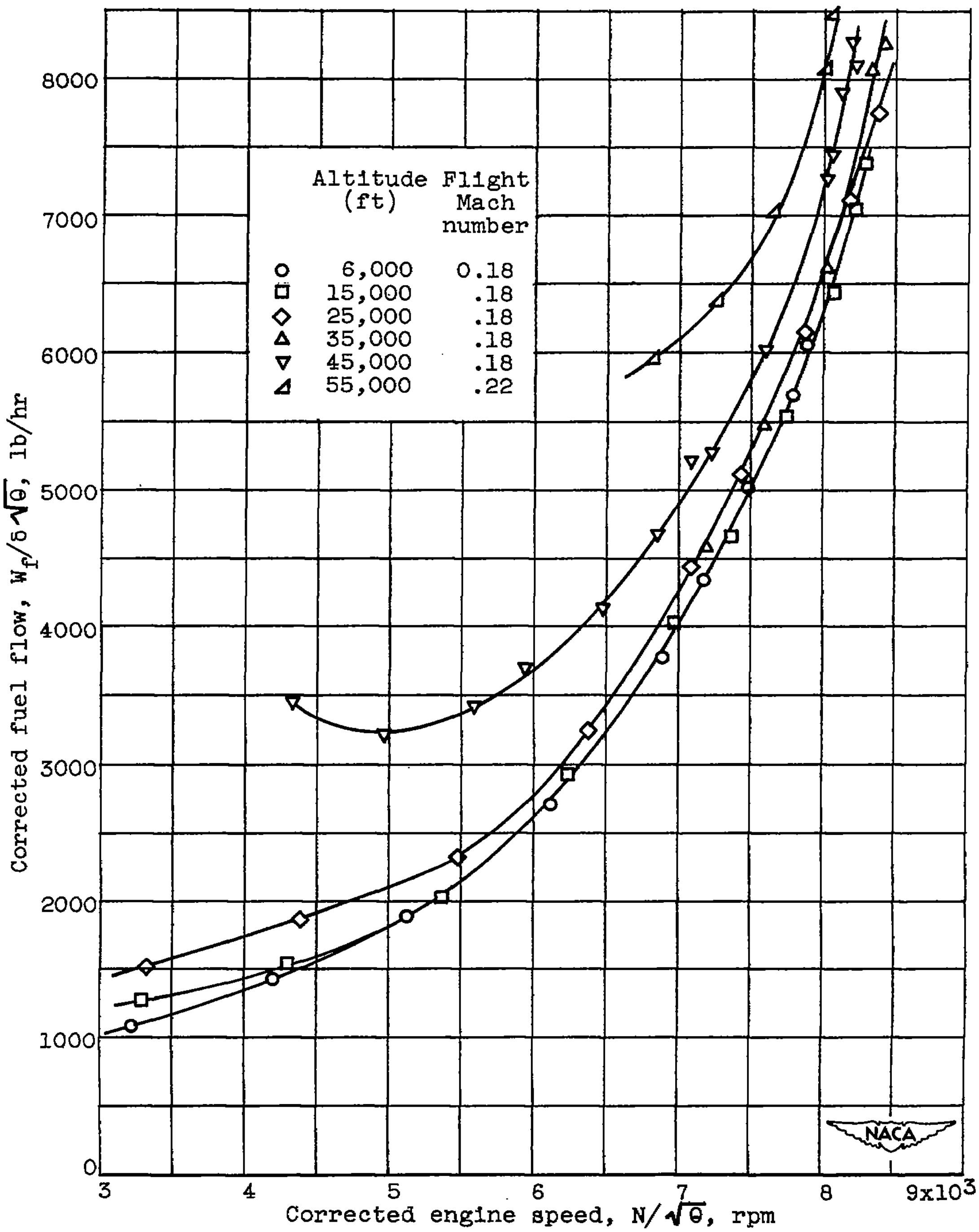
2108



(b) Air flow.

Figure 6. - Continued. Effect of altitude on variation of corrected engine performance with corrected engine speed.

2108



(c) Fuel flow.

Figure 6. - Continued. Effect of altitude on variation of corrected engine performance with corrected engine speed.

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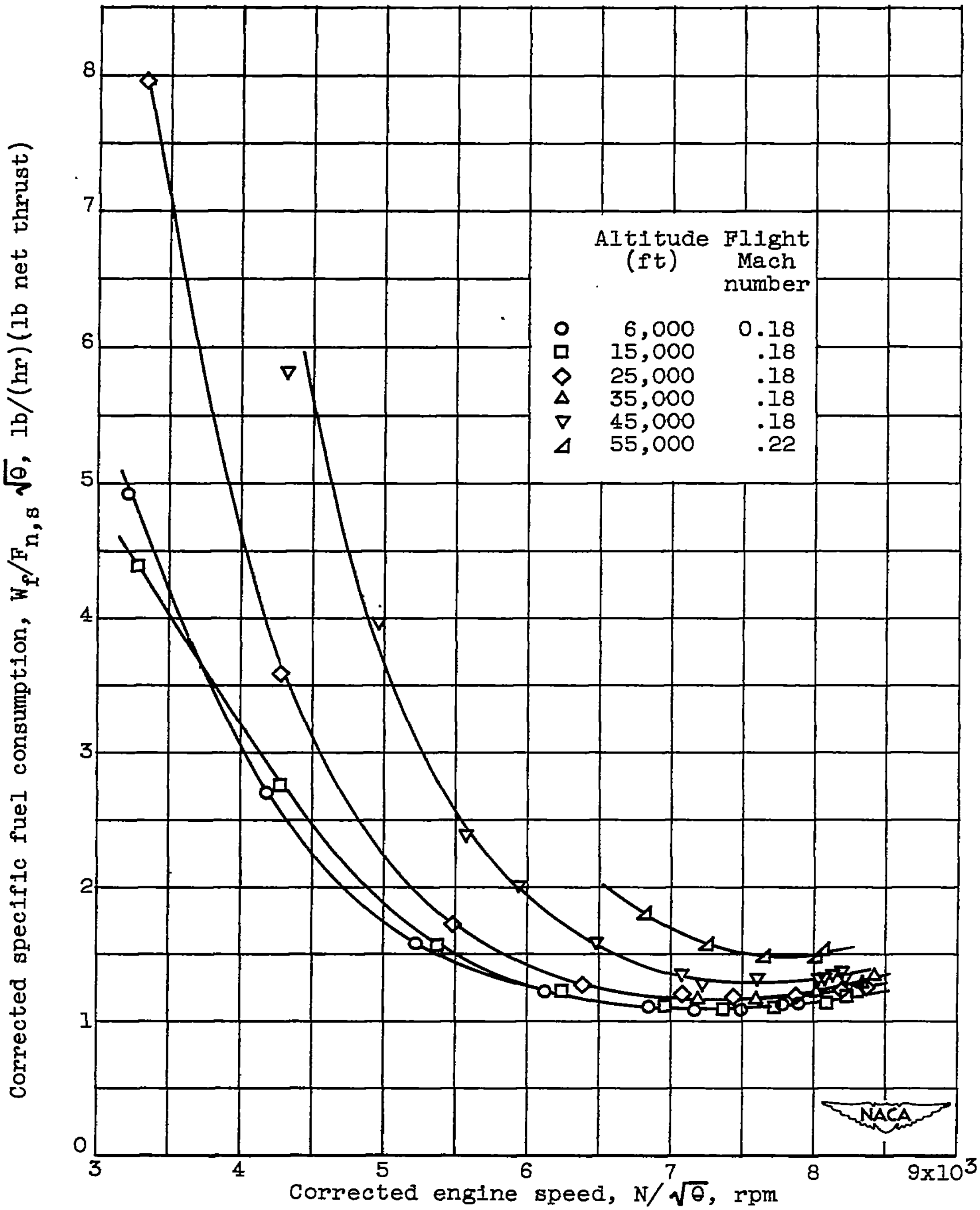
2108

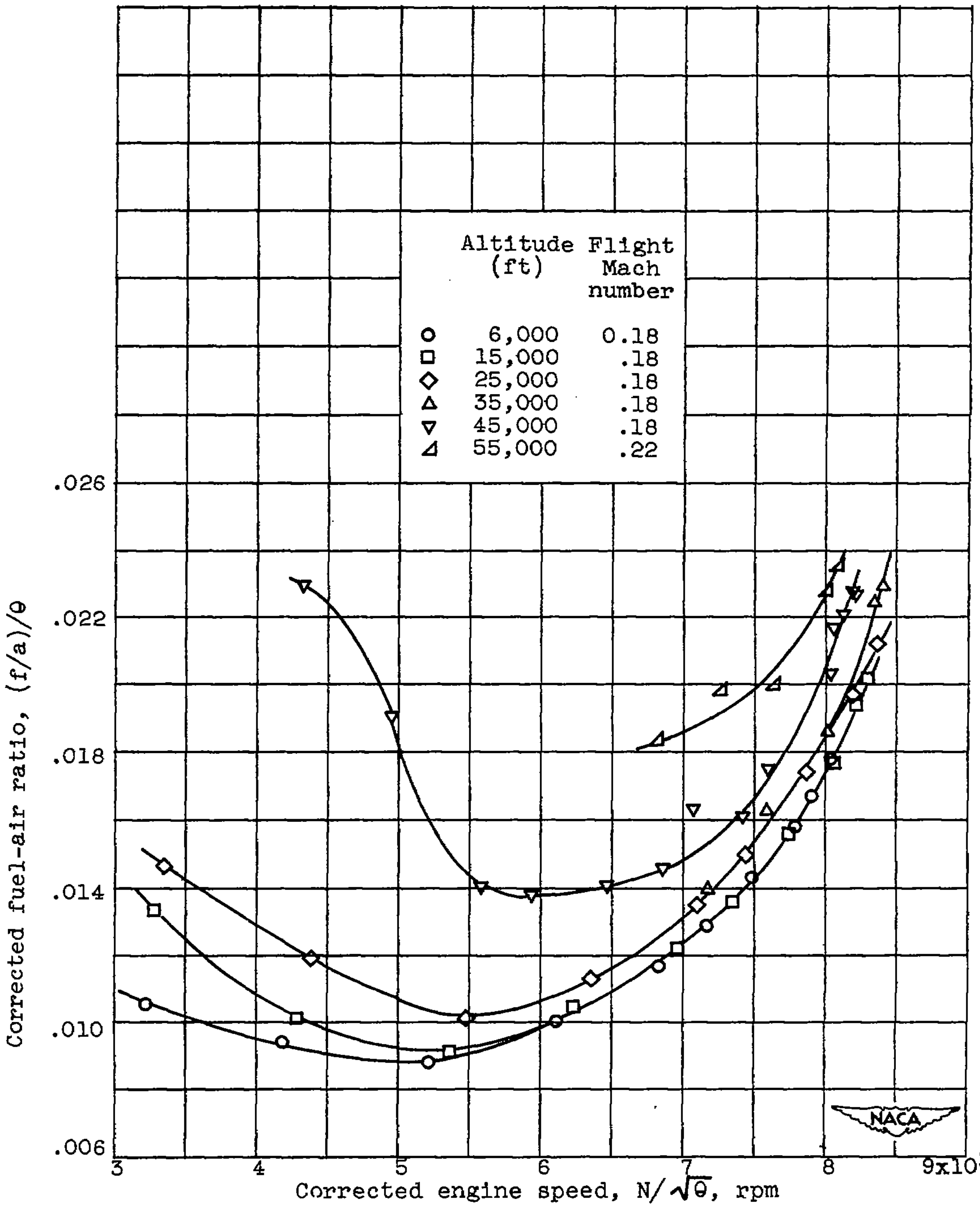


(d) Specific fuel consumption.

Figure 6. - Continued. Effect of altitude on variation of corrected engine performance with corrected engine speed.

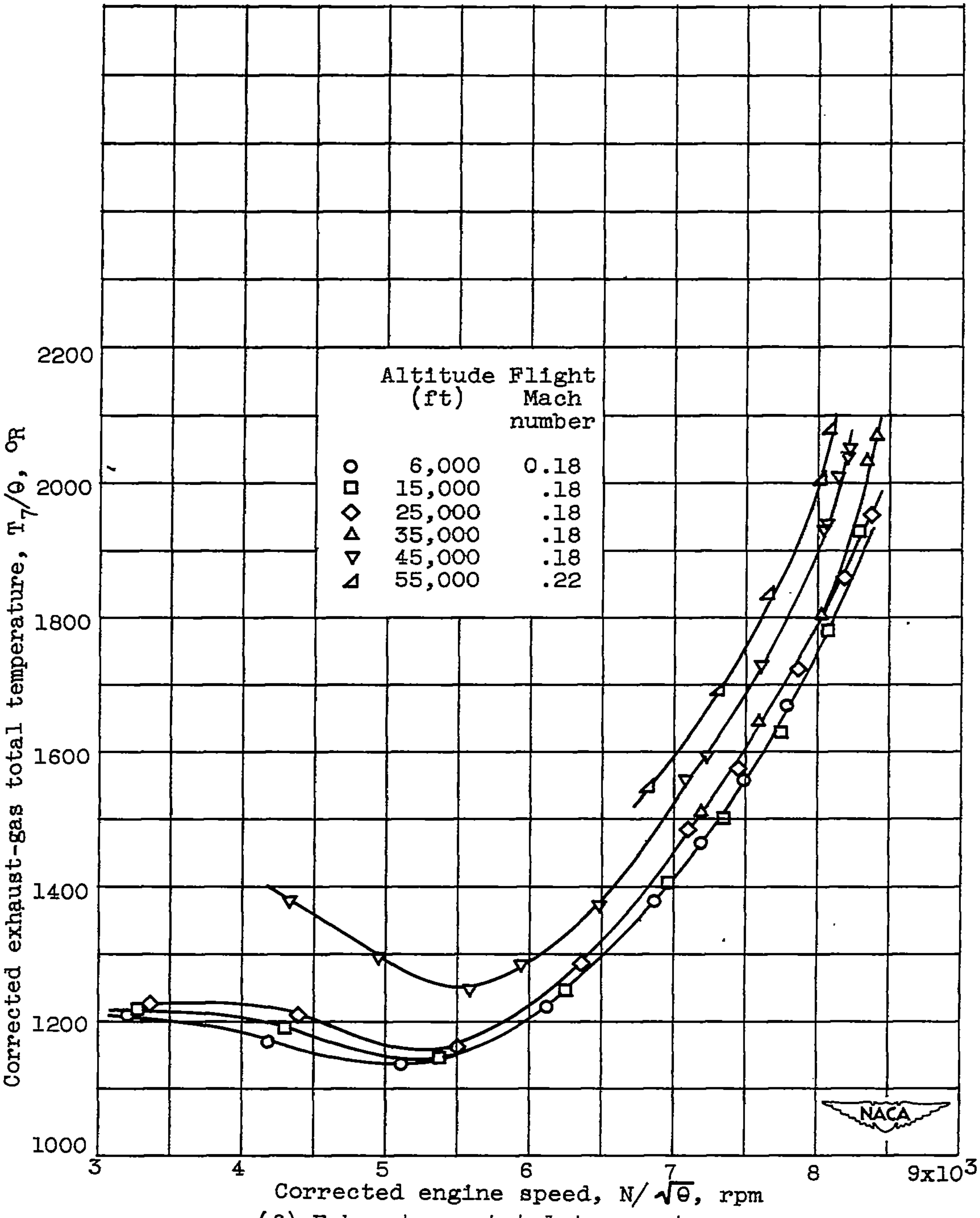
[redacted]

2108



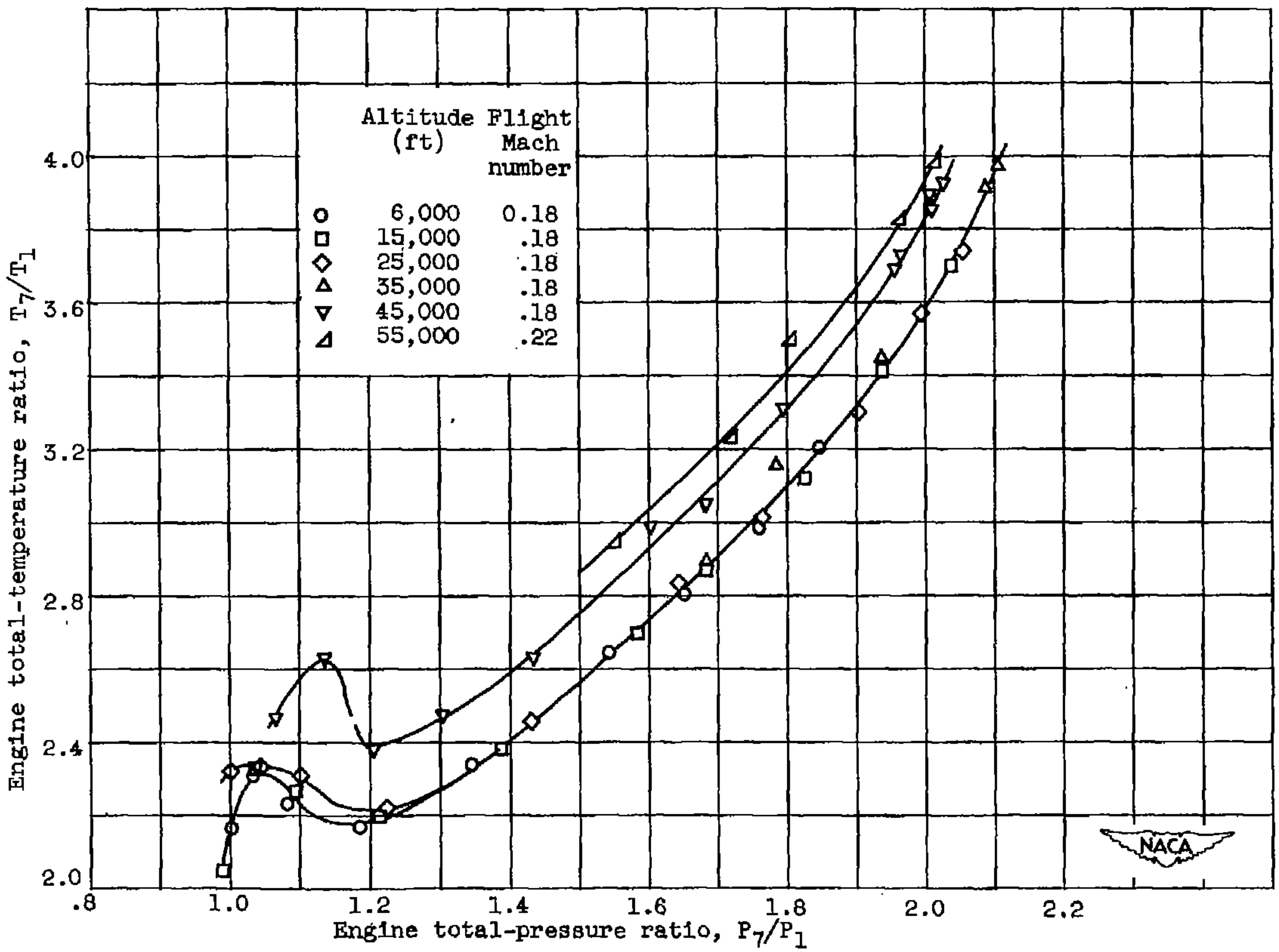
(e) Fuel-air ratio.

Figure 6. - Continued. Effect of altitude on variation of corrected engine performance with corrected engine speed.



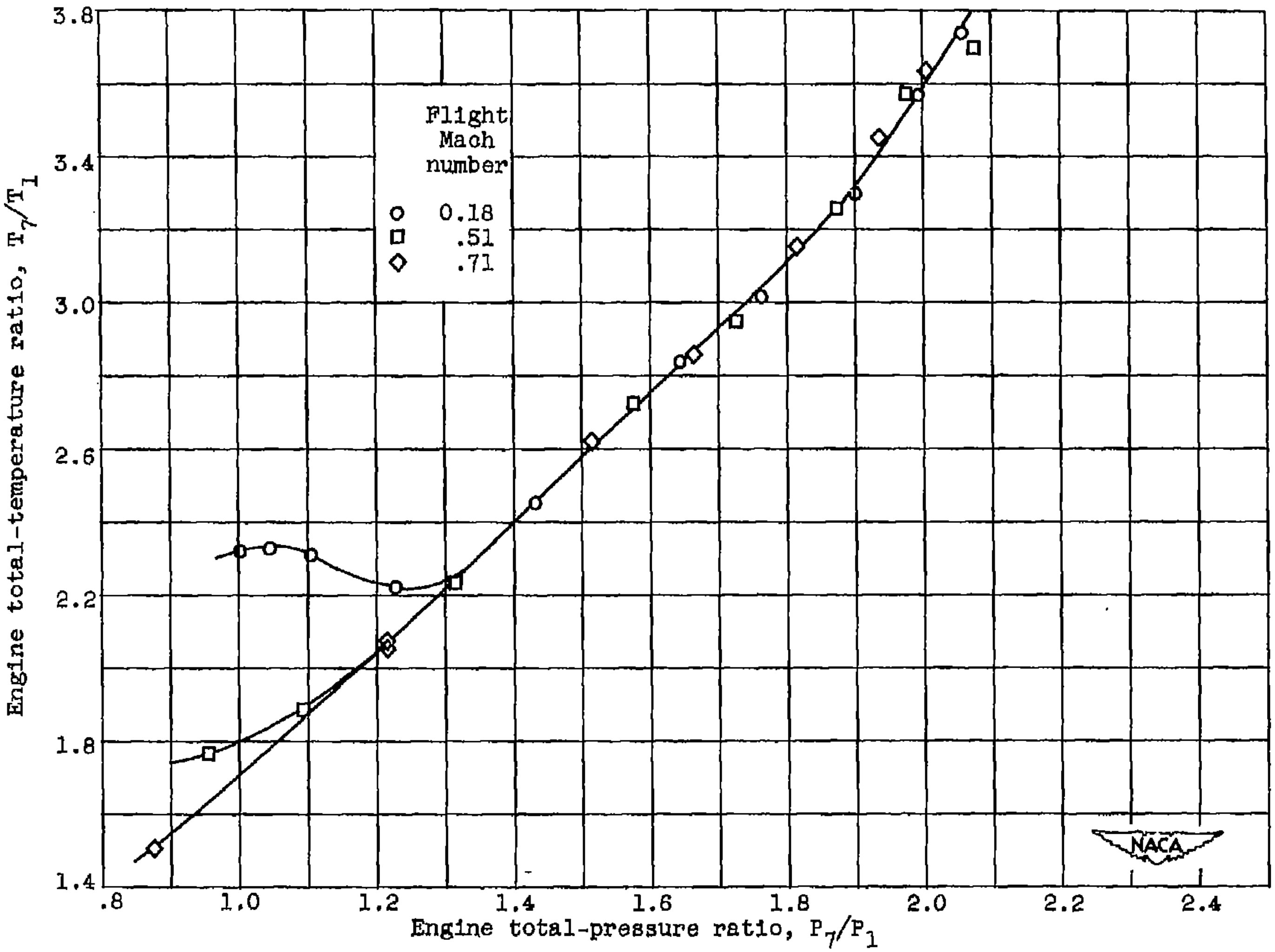
(f) Exhaust-gas total temperature.

Figure 6. - Concluded. Effect of altitude on variation of corrected engine performance with corrected engine speed.



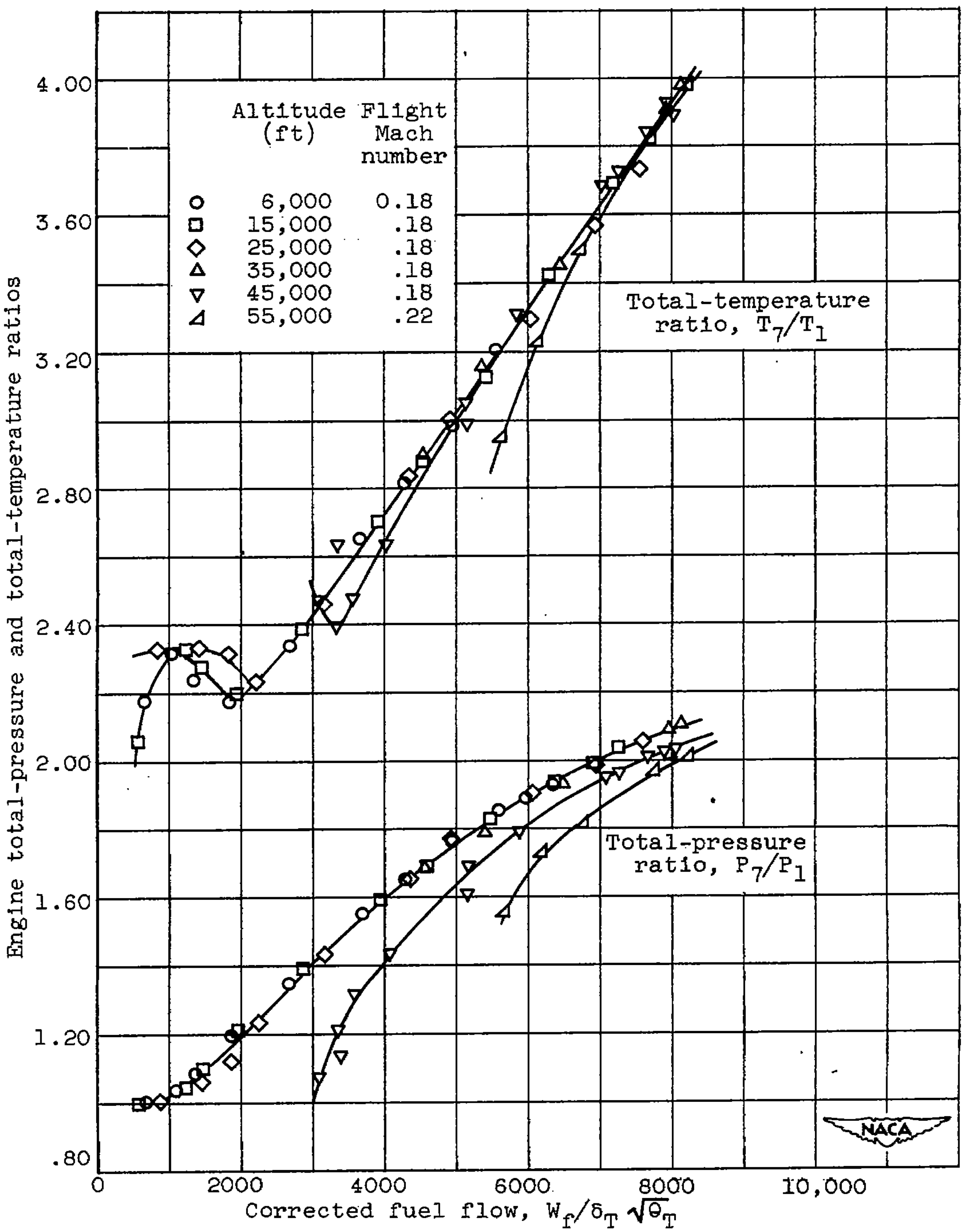
(a) Effect of altitude.

Figure 7. - Variation of engine total-temperature ratio with engine total-pressure ratio.



(b) Effect of flight Mach number at altitude of 25,000 feet.

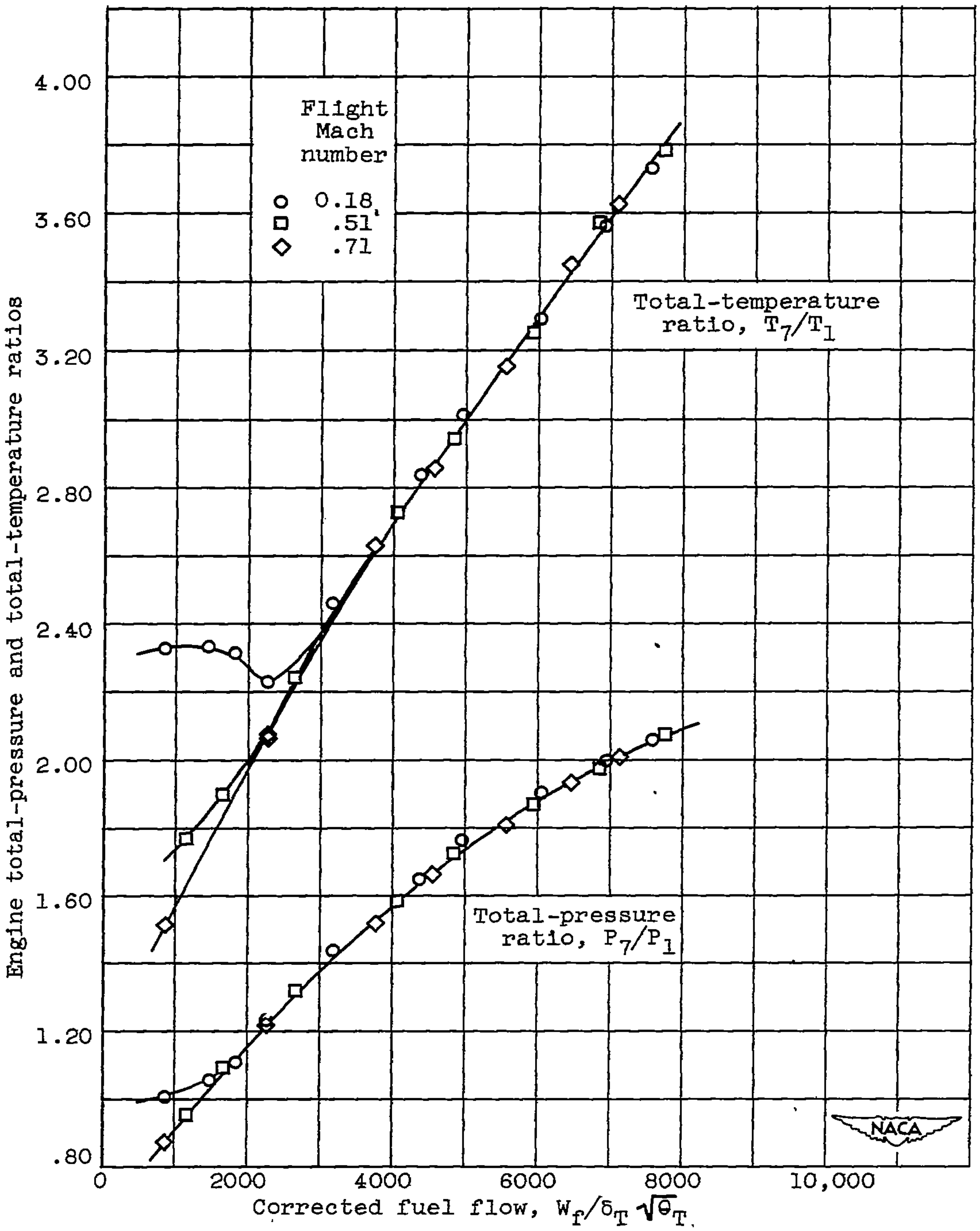
Figure 7. - Concluded. Variation of engine total-temperature ratio with engine total-pressure ratio.



(a) Effect of altitude.

Figure 8. - Variation of engine total-temperature ratio and total-pressure ratio with corrected fuel flow.

[redacted]



(b) Effect of flight Mach number at altitude of 25,000 feet.

Figure 8. - Concluded. Variation of engine total-temperature ratio and total-pressure ratio with corrected fuel flow.

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